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## **MISSILE DATCOM**

### **User's Manual - 2008 Revision**

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## TABLE OF CONTENTS

LIST OF FIGURES.....	iv
LIST OF TABLES.....	v
PREFACE .....	vi
SUMMARY OF MISSILE DATCOM RELEASES .....	vii
1.0 INTRODUCTION .....	1
2.0 PROGRAM CAPABILITIES/INSTALLATION.....	2
2.1 TYPES OF DATA COMPUTED .....	2
2.2 INSTALLATION ON COMPUTER SYSTEMS .....	4
2.2.1 Requirements .....	4
2.2.2 Input/Output .....	5
3.0 INPUT DEFINITION .....	7
3.1 NAMELIST INPUTS .....	8
3.1.1 Namelist FLTCON - Flight Conditions .....	13
3.1.2 Namelist REFQ - Reference Quantities.....	16
3.1.3 Namelist AXIBOD - Axisymmetric Body Geometry.....	18
3.1.4 Namelist ELLBOD - Elliptical Body Geometry .....	26
3.1.5 Namelist PROTUB - Protuberance Geometry .....	30
3.1.6 Namelist FINSETn - Define Fin Set n .....	34
3.1.7 Namelist DEFLCT - Panel Deflection Angles .....	45
3.1.8 Namelist TRIM - Trim Aerodynamics .....	46
3.1.9 Namelist INLET - Axisymmetric and 2-Dimensional Inlet Geometry.....	48
3.1.10 Namelist EXPR - Experimental Data Substitution .....	54
3.2 CONTROL CARD INPUTS .....	56
3.2.1 Control Card - General Remarks.....	56
3.2.2 Control Card Definitions.....	57
3.3 TYPICAL CASE SET-UP .....	62
4.0 OUTPUT DESCRIPTION .....	63
4.1 NOMINAL OUTPUT .....	63
4.1.1 Input Error Checking.....	63
4.1.2 Listing of Case Input Data .....	64
4.1.3 Case Total Configuration Aerodynamic Output Summary.....	65
4.2 PARTIAL OUTPUT.....	66
4.2.1 Geometric Partial Output .....	66
4.2.2 Aerodynamic Partial Output.....	66
4.2.3 Pressure Distribution Data.....	69
4.3 DYNAMIC DERIVATIVES.....	69
4.4 EXTERNAL DATA FILES.....	70
5.0 AERODYNAMIC METHODOLOGY .....	97

## LIST OF FIGURES

Figure 1. Coefficient Axis System.....	3
Figure 2. Axisymmetric Body Geometry Variables .....	25
Figure 3. Blunted and Truncated Nose Variables .....	25
Figure 4. Elliptical Body Variables .....	29
Figure 5. Available Protuberances Shapes.....	32
Figure 6. Sample Missile with Protuberances .....	33
Figure 7. Fin Placement on Body .....	38
Figure 8. Definition of Trailing Edge Flaps .....	39
Figure 9. Fin Numbering and Orientation .....	40
Figure 10. Roll Attitude vs Fin Orientation .....	41
Figure 11. HEX and ARC Airfoil Variables .....	42
Figure 12. USER Airfoil Variables .....	43
Figure 13. Top-Mounted 2-D Inlet/Diverter Geometry .....	50
Figure 14. Side-Mounted 2-D Inlet/Diverter Geometry.....	51
Figure 15. Axisymmetric Inlet/Diverter Geometry .....	52
Figure 16. Geometry Definition for Conformal and Semi-Submerged Inlets .....	53
Figure 17. Typical “Stacked” Case Set-up. ....	62
Figure 18. Example of Missile Datcom Error Messages .....	75
Figure 19. Case Input Listing .....	76
Figure 20. Example of Default Substitutions for Incomplete Case Inputs .....	77
Figure 21. Total Configuration Aerodynamic Output Summary .....	78
Figure 22. Trimmed Output Summary .....	79
Figure 23. Body Geometry Output.....	80
Figure 24. Airfoil Geometry Output.....	81
Figure 25. Fin Geometry Output.....	82
Figure 26. Inlet Geometry Input .....	83
Figure 27. Inlet Geometry Output .....	84
Figure 28. Base-Jet Plume Interaction Output .....	85
Figure 29. Protuberance Output .....	86
Figure 30. Body Alone Aerodynamic Partial Output .....	87
Figure 31. Fin Normal Force and Pitching Moment Partial Output .....	88
Figure 32. Fin Axial Force Partial Output .....	89
Figure 33. Airfoil Section Aerodynamic Partial Output .....	90
Figure 34. Inlet Aerodynamic Partial Output .....	91
Figure 35. Fin Set in Presence of the Body Partial Output.....	92
Figure 36. Fin Set in Presence of the Body Partial Output (continued).....	93
Figure 37. Carryover Interference Factors Partial Output.....	94
Figure 38. Panel Bending Moment Partial Output .....	94
Figure 39. Panel Hinge Moment Partial Output.....	95
Figure 40. Untrimmed Aerodynamic Output (“for003.dat”) .....	95
Figure 41. Dynamic Derivative Output .....	96

## LIST OF TABLES

Table 1. Input/Output Logical Units .....	5
Table 2. Namelist Alphanumeric Constants .....	11
Table 3. NAMELIST FLTCON Variables .....	15
Table 4. NAMELIST REFQ Variables .....	16
Table 5. Equivalent Sand Roughness .....	17
Table 6. Preferred RHR Values .....	17
Table 7. NAMELIST AXIBOD Variables (Option 1 Inputs) .....	21
Table 8. NAMELIST AXIBOD Variables (Option 2 Inputs) .....	22
Table 9. Default values assigned when using AXIBOD Option 2 input scheme. ....	23
Table 10. NAMELIST AXIBOD Base-Jet Plume Interaction Variables .....	23
Table 11. Base-Jet Plume Interaction Parameter Limitations .....	24
Table 12. NAMELIST ELLBOD Variables (Option 1 Inputs) .....	27
Table 13. NAMELIST ELLBOD Variables (Option 2 Inputs) .....	28
Table 14. NAMELIST PROTUB Variables .....	31
Table 15. NAMELIST FINSETn Variables – (Nominal Inputs) .....	37
Table 16. NAMELIST FINSETn Variables- (SECTYP= HEX, ARC inputs) .....	42
Table 17. NAMELIST FINSETn - (SECTYP= USER Inputs) .....	43
Table 18. Airfoil Designation Using the NACA Control Card .....	44
Table 19. NAMELIST DEFLCT Variables .....	45
Table 20. NAMELIST TRIM Variables .....	47
Table 21. NAMELIST INLET Variables .....	49
Table 22. NAMELIST EXPR Variables .....	55
Table 23. Units Required for Flight Condition Variables and Surface Roughness .....	58
Table 24. Magnus derivatives calculated with SPIN Control Card .....	61
Table 25. Dictionary listing for the FOR020.DAT file .....	71
Table 26. Dictionary listing for the FOR021.DAT file .....	72
Table 27. Dictionary listing for the FOR042.CSV file .....	73
Table 28. Dictionary listing for the FOR043.CSV file .....	74
Table 29. Body Alone Aerodynamic Methodology .....	98
Table 30. Body Alone Subroutines .....	99
Table 31. Fin Alone Aerodynamic Methodology .....	100
Table 32. Fin Alone Subroutines .....	100
Table 33. Inlet Aerodynamic Methodology .....	101
Table 34. Inlet Subroutines .....	101
Table 35. Body-Fin Synthesis Aerodynamic Methodology .....	102
Table 36. Body-Fin Synthesis Subroutines .....	102

## PREFACE

This report was prepared by the System Simulation and Development Directorate of the US Army Aviation and Missile Research, Development and Engineering Center (AMRDEC) in conjunction with the Air Vehicles Directorate of the Air Force Research Laboratory (AFRL/RB), Wright-Patterson AFB, Ohio. It documents the August 2008, FORTRAN 90 version of Missile Datcom. The development of the original FORTRAN 77 version of Missile Datcom was performed by the McDonnell Douglas Corporation, St. Louis, Missouri. This report supersedes AFRL-VA-WP-TR-1998-3009 that documents the 1997 Datcom revision and WL-TR-93-3043 that documents the 1993 revision of Missile Datcom.

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The Air Vehicles Directorate is planning to continue development of Missile Datcom. This development is dependent to a large extent on user feedback. Questions about the program or suggestions for future improvements to the program should be directed to Mr. William Blake, AFRL/RB, Wright-Patterson AFB, Ohio 45433, phone (937) 255-6764

## SUMMARY OF MISSILE DATCOM RELEASES

Contract	Investigator (Govt engineer)	Documentation	Release	Rev.	Capability added
F33615-80-C-3605 (McDonnell Douglas)	S.R. Vukelich (J.E. Jenkins)	AFWAL TR-81-3130			Feasibility study only Recommended methods, code structure
F33615-81-C-3617 (McDonnell Douglas)	S.R. Vukelich (J.E. Jenkins)		1	12/84	Axisymmetric bodies Two fin sets with up to four fins each Automatic configuration trim
Same	S.R. Vukelich (J.E. Jenkins)		2	11/85	Elliptical bodies, Inlets at supersonic speeds Dynamic derivatives, exp data substitution Four fin sets with up to eight fins each
Same	S.L. Stoy (J.E. Jenkins)	AFWAL-TR-86-3091 (ADA 211086, 210128)	3	12/88	Expanded data substitution Configuration incrementing
None	(W.B. Blake)		4	7/89	Expanded body dynamic derivatives
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F33657-89-D-2198 (McDonnell Douglas)	K.A. Burns (J.W. Herrmann)	WL-TR-93-3043 (ADA 267447)	6	6/93	UNIX workstation, PC compatibility Trailing edge flaps, Folding fins Semi-submerged inlets
None	(W.B. Blake)	AFRL-TR-1998-3009 (ADA-344707)	7	5/97	Fortran 90 compatibility Expanded dynamic derivatives Revised body-fin upwash, Modified base drag Modified fin longitudinal center of pressure
None	(W.B. Blake)	AIAA-1999-4258	8	09/02	High AOA, airfoil, friction drag corrections
None	M. Underwood (L. Auman, Army) (W.B. Blake, AF)	AIAA-2003-3668 AIAA-2005-4833 AIAA-2005-4971	9	01/06	Code clean-up and restructuring Cambered body capability Revised body drag (bluntness, high AOA)
DAAH01-02-C-R170 W31P4Q-07-D-0001	M. Underwood C. Rosema (L. Auman, Army)	AIAA-2007-3936 AIAA-2007-3937	10	07/07	Rolling moment for elliptical bodies Improvements to TE Flap control increments Improvements for low aspect ratio fins
W31P4Q-07-D-0001	M. Underwood Joshua Doyle C. Rosema (L. Auman, Army)	AIAA-2009-0907 This report	11	08/08	Nine fin sets with up to 8 fins each Revised body drag Moment contribution from protuberances



## 1.0 INTRODUCTION

In missile preliminary design it is necessary to quickly and economically estimate the aerodynamics of a wide variety of missile configuration designs. Since the ultimate shape and aerodynamic performance are so dependent upon the subsystems utilized, such as payload size, propulsion system selection and launch mechanism, the designer must be capable of predicting a wide variety of configurations accurately. The fundamental purpose of Missile Datcom is to provide an aerodynamic design tool which has the predictive accuracy suitable for preliminary design, and the capability for the user to easily substitute methods to fit specific applications.

## 2.0 PROGRAM CAPABILITIES/INSTALLATION

The computer code is capable of addressing a wide variety of conventional missile designs. For the purposes of this document, a conventional missile is one which is comprised of the following:

- An axisymmetric or elliptically-shaped body.
- One to nine fin sets located along the body between the nose and base. Each fin set can be comprised of one to eight identical panels attached around the body at a common longitudinal position. Each fin may be deflected independently, as an all moving panel or as a fixed panel with a plain trailing edge flap.
- An airbreathing propulsion system.

To minimize the quantity of input data required, commonly used values for many inputs are assumed as defaults. However, all program defaults can be overridden by the user in order to more accurately model the configuration of interest.

### 2.1 TYPES OF DATA COMPUTED

#### 2.1.1 Aerodynamics

The program computes the following aerodynamic parameters as a function of angle of attack for each configuration:

$C_N$	Normal Force Coefficient (body axis)
$C_L$	Lift Coefficient (wind axis)
$C_M$	Pitching Moment Coefficient (body axis)
$X_{cp}$	Center of Pressure in calibers from the moment reference center
$C_A$	Axial Force Coefficient (body axis)
$C_D$	Drag Coefficient (wind axis)
$C_Y$	Side Force Coefficient (body axis)
$C_n$	Yawing Moment Coefficient (body axis)
$C_l$	Rolling Moment Coefficient (body axis)
$C_{N\alpha}$	Normal force coefficient derivative with angle of attack
$C_{m\alpha}$	Pitching moment coefficient derivative with angle of attack
$C_{Y\beta}$	Side force coefficient derivative with sideslip angle
$C_{n\beta}$	Yawing moment coefficient derivative with sideslip angle (body axis)
$C_{l\beta}$	Rolling moment coefficient derivative with sideslip angle (body axis)

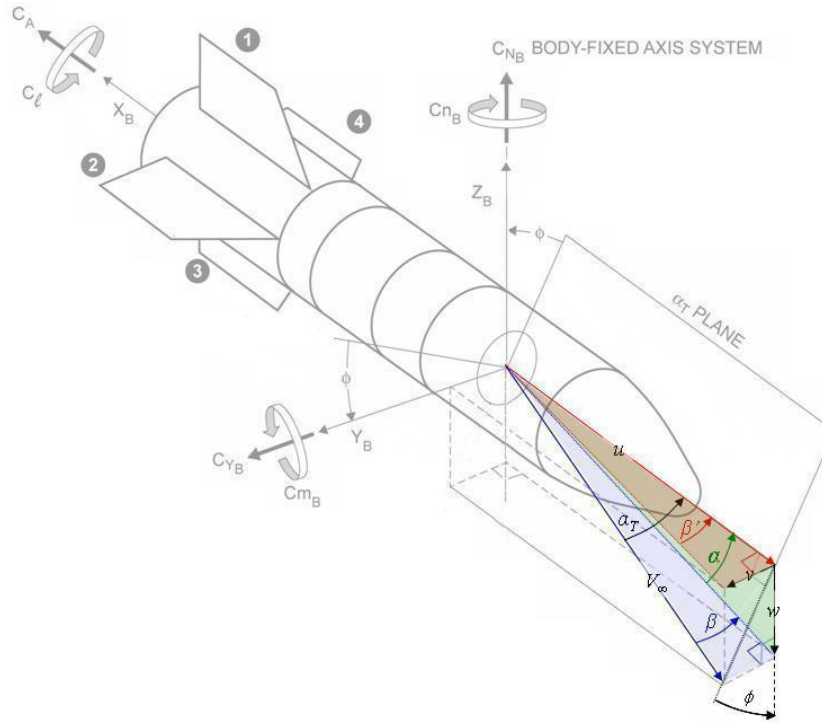


Figure 1. Coefficient Axis System

For body alone or body plus fin combinations, the following parameters are also computed, all in the body axis system:

$CM_q$	Pitching moment coefficient derivative with pitch rate
$CN_q$	Normal force coefficient derivative with pitch rate
$CA_q$	Axial force coefficient derivative with pitch rate
$CM_{\dot{\alpha}}$	Pitching moment derivative with rate of change of angle of attack
$CN_{\dot{\alpha}}$	Normal force derivative with rate of change of angle of attack
$Cl_p$	Rolling moment coefficient derivative with roll rate
$Cn_p$	Yawing moment coefficient derivative with roll rate
$CY_p$	Side force coefficient derivative with roll rate
$Cl_r$	Rolling moment coefficient derivative with yaw rate
$Cn_r$	Yawing moment coefficient derivative with yaw rate
$CY_r$	Side force coefficient derivative with yaw rate

The output units of the aerodynamic derivatives can be in degrees or radians and is controlled by the DERIV control card. This control card has no effect on input angles, input angles are always specified in degrees. Partial output results, which detail the components used in the calculations, are also optionally available using the PART and/or BUILD control cards.

The drag force (and drag coefficient) is different between the wind and stability axes systems if the missile body is at a sideslip angle ( $\beta$ ) to the wind. However, wind axis drag and stability axis drag are the same at zero sideslip. In Missile Datcom, drag force methods are assumed to be in the stability axes system and axial force methods are assumed to be in the body axes system unless otherwise noted.

The program has the capability to perform a static trim of the configuration, using any fin set for control with fixed incidence on the other sets. The two types of aerodynamic output available from the trim option are as follows:

- Untrimmed data - Each of the aerodynamic force and moment coefficients are printed in a matrix, which is a function of angle of attack and panel deflection angle. This output is optional.
- Trimmed data - The trimmed aerodynamic coefficients, and trim deflection angle, are output as a function of angle of attack.

### 2.1.2 Geometry

All components of the configuration have their physical properties calculated and output for reference if requested. All data is supplied in the user selected system of units. The reference area and reference length are user defined.

## 2.2 INSTALLATION ON COMPUTER SYSTEMS

This section details the steps necessary to make the computer code functional on the user's computer system. Although conversion of the program can be easily accomplished by someone with a good understanding of the FORTRAN language, it is highly recommended that someone familiar with the computer operating system be consulted.

### 2.2.1 Requirements

In order for the Missile Datcom code to be successfully implemented on the user's computer system, there are three requirements which must be met, as follows:

- Language - The code is written in FORTRAN 77 but is also compatible with FORTRAN 90/95.
- Namelist - The code has been designed with an internal FORTRAN NAMELIST emulator to allow the input and output (I/O) to be handled by namelist variables. This is an exception to Standard FORTRAN but with the emulator as part of the code the program will run under Standard FORTRAN. The code is not easily converted to fixed field, rather than namelist input.
- I/O Scratch Files - The code uses the logical file units listed in **Table 1**. All file units are accessed using formatted reads and writes. File units 2 and 8 are used internally; file units 3, 5, 6, 9, 10, 11, 12, 20, 21, 22, 42 and 43 transfer data between the user and the code.

### 2.2.2 Input/Output

Fourteen file units are used by the program. They are used as follows:

Table 1. Input/Output Logical Units

<u>Unit</u>	<u>Name</u>	<u>Usage</u>	<u>Subroutine</u>
2	for002.dat	Namelists for the input "case" are read from unit 8 and written to unit 2.	READIN
3	for003.dat	Plot file of aerodynamic data, written at user request (using PLOT card) to unit 3	PLOT3 PLTTRM PLTUT9
5	for005.dat	User input file read from unit 5	CONERR
6	for006.dat	Program output file written to unit 6	PRINTS PRIIOM
8	for008.dat	User input cards read from unit 5 are written to unit 8 after they have been checked for errors.	CONERR
9	for009.dat	Body geometry data, written at user request to unit 9	SBODY
10	for010.dat	Body pressure coefficient data at angle of attack, written at user request to unit 10 when using PRESSURES card.	SOSE VANDYK HYPERS
11	for011.dat	Fin pressure coefficient data, written at user request to unit 11 when using PRESSURES card	FCAWPF
12	for012.dat	Body pressure coefficient and local Mach number at zero angle of attack, written at user request to unit 12 when using PRESSURES card	SOSE
20	for020.dat	Total configuration force and moment coefficient data, damping derivatives and flight conditions, written at user request (using PLOT card). File is formatted for use with software developed with Adaptive Modeling Language (AML).	PLOT20
21	for021.dat	Total configuration force and moment coefficient data, damping derivatives, flight conditions and control deflections, written at user request (using PLOT card). File is formatted for use with the Aviator Visual Design Simulator (AVDS).	PLOT21
22	for022.dat	Configuration geometry file compatible with the commercial software package Tecplot. Only geometry for the body and fins are printed. No geometry is provided for inlets or probuterances.	TECGRD
42	for042.csv	All standard data written in rows and columns with headers	PLOT42
43	for043.csv	Fin data written in rows and columns with headers	PLOT43

As the number of output columns change depending on what input flags the user has set, it is recommended that any user developed software that reads this file should key off the column header names and not specific column locations.

The program is run in a “batch” mode. The user prepares an input file in accordance with the rules given in **Section 3** of this report. This file must be renamed “for005.dat” prior to program execution. The program is then executed by double clicking the executable file. The program then executes and creates the output files requested by the case inputs. The primary output files, “for006.dat” and “for042.csv” are always written. A complete discussion of what is contained in this output file is given in **Section 4** of this report. The optional plot, geometry, and pressure distribution output files are written only if requested.

### 3.0 INPUT DEFINITION

Inputs to the program are grouped by "case". A "case" consists of a set of input cards which define the flight conditions and geometry to be run. Provisions are made to allow multiple cases to be run. The successive cases can either incorporate the data of the previous case (using the input card SAVE) or be a completely new configuration design. The SAVE feature, for example, permits the user to define a body and wing (or canard) configuration in the first case and vary the tail design for subsequent cases.

The scheme used to input data to the computer program is a mixture of namelist and control cards. This combination permits the following:

- Inputs are column independent and can be input in any order.
- All numeric inputs are related to mnemonic (variable) names.

The program includes an error checking routine which scans all inputs and identifies errors. The error checking has been significantly increased in the 7/07 and 8/08 releases. This process is a single-pass error checking routine; errors are identified in a single "run". In addition, the program checks for necessary valid inputs, such as a non-zero Reynolds number. In general the code will skip over cases with errors and provide an error message to the user.

Flexibility has been maintained for all user inputs and outputs. The following summarize the program generality available:

- The unit system can be English or Metric and is controlled by the DIM card. The four options are feet, inches, meters, and centimeters. DIM FT and DIM IN toggle the English unit system while DIM M and DIM CM toggle the Metric unit system. Missile geometry inputs are specified by the chosen length following DIM while other input flight conditions and reference quantities such as altitude, velocity, pressure, and temperature are associated with the unit system chosen rather than actual length. The default is DIM FT. Refer to **Table 23** for a detailed listing of corresponding units.
- Output derivatives can be expressed in degree or radian measure, and is specified via the DERIV control card. Degree measure is the default.
- The body geometry can be defined either by shape type or by surface coordinates.
- The airfoil can be user defined, NACA, or supersonic shaped sections. The NACA sections are defined using the NACA designation. A hexagonally shaped supersonic section is the default.
- The configuration can be run at a fixed sideslip angle and varying body angle of attack, or a fixed aerodynamic roll angle and varying total angle of attack.
- The flight conditions can be user defined, or set using a Standard Atmosphere model. The capability to define wind tunnel test conditions as the flight conditions is also available.

### 3.1 NAMELIST INPUTS

The required program inputs use FORTRAN namelists. Missile Datcom is similar to other codes which use the namelist input technique, but differs as follows:

- Namelist inputs are column independent, and can begin in any column including the first. If a namelist is continued to a second card, the continued card must leave column 1 blank. Also, the card before the continued card must end with a comma. The last usable column is number 79 if column 1 is used, and column 80 if column 1 is blank.
- The same namelist can be input multiple times for the same input case, **however this is NOT a recommended practice and should be avoided.**

The three namelist inputs

```
$REFQ      SREF=1.,$  
$REFQ      LREF=2.,$  
$REFQ      ROUGH=0.001,$
```

are equivalent to

```
$REFQ      SREF=1.,LREF=2.,ROUGH=0.001,$
```

- If the SAVE card is used, the total number of namelists read, including repeat occurrences of the same namelist name from all previous SAVES, must not exceed 300.
- The last occurrence of a namelist variable in a case is the value used for the calculations.

The three namelist inputs

```
$REFQ      SREF=1.,$  
$FLTCON    NMACH=2.,MACH=1.0,2.0,$  
$REFQ      SREF=2.,$
```

are equivalent to

```
$REFQ      SREF=2.,$  
$FLTCON    NMACH=2., MACH=1.0, 2.0,$
```

- Certain variables may be input as arrays instead of single values, such as ALPHA. If the array list is too long (80 columns) to fit on one line, it must be continued on the following line. This can be done by repeating the variable name with the array index of the first continued value, or continuing to the next line with the next value. For example:

```
$FLTCON  
  NALPHA=20., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,
```



ALPHA(12)=22.,24.,28.,32.,36.,40.,44.,48.,52.,  
 NMACH=5., MACH=0.2,0.8,1.5,2.0,3.0, ALT=0.,10000.,  
 ALT(3)=20000.,30000.,40000.,\$

or

\$FLTCON  
 NALPHA=20., ALPHA=0.,2.,4.,6.,8.,10.,12.,14.,16.,18.,20.,  
 22.,24.,28.,32.,36.,40.,44.,48.,52.,  
 NMACH=5., MACH=0.2,0.8,1.5,2.0,3.0, ALT=0.,10000.,  
 20000.,30000.,40000.,\$

- The namelists can be input in any order.
- Only those namelists required to execute the case need be entered.
- Certain alphanumeric constants are permitted. They are summarized in **Table 2**.
- Note the constant UNUSED can be used in the input deck, but it cannot be guaranteed to reset the value truly to the value used within the code (i.e., the read in value may not be truly .EQ. to UNUSED within the code due to round-off error). It is therefore recommended that UNUSED be avoided. **The recommended way of resetting a variable within a namelist input is to use the DELETE command on the namelist, then set the values as desired.**

All Missile Datcom namelist inputs are either real numbers or logical constants. Integer constants will produce a nonfatal error message from the error checking routine and should be avoided. **ALL namelist and variable names must be input in CAPITAL LETTERS.** This also applies to numerical values input in "E" format, i.e. REN=6.0E06 is acceptable, while REN=6.0e06 is not.

The namelist names have been selected to be mnemonically related to their physical meaning. Each component of the configuration requires a separate namelist input.

\$FLTCON	to define the flight conditions
\$AXIBOD or \$ELLBOD	to define the body
\$FINSET1	to define the first fin set
\$FINSET2	to define the second fin set
\$FINSET3	to define the thirdfin set
\$FINSET4	to define the fourth fin set
\$FINSET5	to define the fifth fin set
\$FINSET6	to define the sixth fin set
\$FINSET7	to define the seventh fin set
\$FINSET8	to define the eighth fin set
\$FINSET9	to define the ninth fin set
\$PROTUB	to define protuberance option inputs
\$INLET	to define inlet geometry

The following namelists are optional since defaults exist for all inputs:

\$REFQ	to define the reference quantities
\$DEFLCT	to define the panel incidence (deflection angles)
\$TRIM	to define a trim case
\$EXPR	to define experimental input data

Defaults for all namelists should be checked to verify the configuration being modeled does not include an unexpected characteristic introduced by a default.

Table 2. Namelist Alphanumeric Constants

<b>NAMELIST</b>	<b>PERMITTED ALPHANUMERIC CONSTANTS</b>	<b>CONVERTED VALUE</b>
(ALL)	UNUSED	1.E-30 (not recommended)
REFQ	TURB	0.
	NATURAL	1.
AXIBOD or ELLBOD	CONICAL	0.
	CONE	0.
	OGIVE	1.
	POWER	2.
	HAACK	3.
	KARMAN	4.
PROTUB	VCYL	1.
	HCYL	2.
	LUG	3.
	SHOE	4.
	BLOCK	5.
	FAIRING	6.
FINSET <sub>n</sub>	HEX	0.
	NACA	1.
	ARC	2.
	USER	3.
INLET	2DTOP	3.
	2DSIDE	1.
	AXI	2.
EXPR	BODY	1.
	F1	2.
	F2	3.
	F3	4.
	F4	5.
	BF1	6.
	BF12	7.
	BF123	8.
	BF1234	9.

The following sections describe each of the namelist inputs. Each section is accompanied by a figure which summarizes the input variables, their definitions, and units. Since the system of units can be optionally selected, the column "Units" specifies the generic system of units as follows:

L	Units of length; feet, inches, centimeters or meters
F	Units of force; pounds or Newtons
deg	Units of degrees; if angular, in angular degrees; if temperature, either degrees Rankine or Kelvin
sec	Units of time in seconds

Exponents are added to modify the above. For example,  $L^2$  means units of length squared, or area. Combinations of the above are also used to specify other units. For example,  $F/L^2$  means force divided by area, which is a pressure.

Since it is difficult to discern the difference between the number zero "0" and the alphabetic letter "O", it should be noted that none of the namelist or namelist variable names contain the number zero in them. In general, the number zero and the letter "O" are not interchangeable unless so stated.

The program ascertains the configuration being modeled by the presence of each component namelist, even if no data is entered. The following rules for namelist input apply:

- Do not include a namelist unless it is required. Once read, the presence of a namelist (and, hence, a configuration component) can only be removed using the DELETE control card in a subsequent case. Simply setting all variables to their initialized values will not remove the configuration component.
- Do not include a variable within a namelist unless it is required. Program actions are often determined from the number and types of input provided.
- Do not over-specify the geometry. In the 7/07 and 8/08 releases, over-specified geometry should always cause an error and STOP the code.

### 3.1.1 Namelist FLTCON - Flight Conditions

This namelist defines the flight conditions to be run for the case. In the 7/07 and 8/08 releases, the program is limited to no more than 100 angles of attack and 20 Mach number/Altitude combinations per case at a fixed sideslip angle, aerodynamic roll angle, and panel deflection angle. Therefore, a "case" is defined as a fixed geometry with variable Mach number/Altitude and angles of attack.

The inputs are given in **Table 3**. There are two ways in which the aerodynamic pitch and yaw angles can be defined:

- Input ALPHA and BETA. If BETA is input and PHI is not, it is assumed that the body axis angles of attack ( $\alpha$ ) and sideslip angles ( $\beta$ ) are defined.

**NOTE: The 7/07 and all prior releases defined BETA as  $\beta = \tan^{-1}(v/u)$ , which is shown as  $\beta'$  in Figure 1. The 8/08 release defines BETA as  $\beta = \sin^{-1}(v/V_o)$  which is shown as  $\beta$  in Figure 1. Equations for converting between the two definitions are given in Table 3.**

- Input ALPHA and PHI. If PHI is input and non-zero, it is assumed that ALPHA is the total angle of attack ( $\alpha$ ) and PHI is the aerodynamic roll angle ( $\phi$ ).

**NOTE: Input ALPHA, BETA and PHI is NOT allowed. The 7/07 and 8/08 releases will generate an error message and STOP execution of the code.**

NOTE: Missile Datcom will not run if two or more identical ALPHA values are defined in the input file.

As a minimum the following variables must be defined:

NALPHA	number of angles of attack to run (NALPHA $\geq$ 2)
ALPHA	angle of attack schedule (matching NALPHA)
NMACH	number of Mach numbers or speeds (NMACH $\geq$ 1)
MACH or VINF	Mach number or speed schedule (matching NMACH)

**NOTE: The ALT, REN, TINF and PINF data must correspond to the MACH or VINF inputs. The ALPHA and MACH (or VINF) dependent data can be input in any order; the code will sort the data into ascending order. If MACH or VINF is sorted by Missile Datcom, the corresponding REN, ALT, TINF, and PINF will also be reordered to maintain the correct pairing with MACH or VINF.**

Missile Datcom requires Mach number and Reynolds number to operate. The program can calculate these from several combinations of free-stream input conditions.

- |                        |   |
|------------------------|---|
| 1. MACH and REN        | (no computations required – primary input)                            |
| 2. MACH and ALT        | (Reynolds number is computed using the Standard Atmosphere)           |
| 3. MACH, TINF and PINF | (Reynolds number is computed)   |
| 4. VINF and ALT        | (Reynolds and Mach number are computed using the Standard Atmosphere) |
| 5. VINF, TINF and PINF | (Reynolds and Mach number are computed)                               |
| 6. VINF, TINF and REN  | (Mach number is computed)   |

User supplied data will take precedence over program calculations. Hence, the user can override any default or Standard Atmosphere calculation.

**NOTE: Previous versions of Missile Datcom used ALT=0 (sea-level) as a default condition. This default has been eliminated in the 7/07 version as it was found to be a source of user error. The user must specify the flight conditions, however the user may mix and match the data if he chooses. For example:**

```
VINF(1) = 300., ALT(1) = 30000.,  
MACH(2) = 2., ALT(2) = 0.,
```

**or**

```
VINF(1) = 300., TINF(1) = 288., REN(1) = 1.E+6,  
MACH(2) = 2., REN(2) = 2000000.,
```

**are valid entries.**

## NAMELIST FLTCON

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS**	DEFAULT															
NALPHA	-	Number of angles of attack (must be > 1)	-	-															
ALPHA	100	Angle of attack or total angle of attack, $\alpha = \tan^{-1}(w/u)$ or $\alpha_T = \tan^{-1}(\sqrt{(v^2 + w^2)/u})$ (see Figure 1)	Deg	-															
BETA***	-	Sideslip angle, $\beta = \tan^{-1}(v/V_o)$ , (see Fig. 1)	Deg	0.															
PHI***	-	Aerodynamic roll angle $\phi = \tan^{-1}(v/w)$ , (see Figure 1)	Deg	0.															
NMACH	-	Number of Mach numbers or velocities	-	-															
MACH*	20	Mach numbers	-	-															
ALT*	20	Altitudes	L	0.															
REN*	20	Reynolds numbers per unit length	1/L	-															
VINF*	20	Freestream velocities	L/sec	-															
TINF*	20	Freestream static temperatures	Deg	-															
PINF*	20	Freestream static pressures	F/L <sup>2</sup>	-															
<div>Note:<div>*Any of the following combinations satisfy the minimum requirements for calculating atmospheric conditions (Mach and Reynolds number):<div>1. MACH and REN</div><div>2. MACH and ALT</div><div>3. MACH and TINF and PINF</div><div>4. VINF and ALT</div><div>5. VINF and TINF and PINF</div><div>6. VINF and TINF and REN</div></div><div>**Lengths are in feet for English units and meters for metric units.</div><div>***PHI and BETA can NOT be used in the same case.</div></div> <table><tr><td><math>(\alpha_T, \phi) \text{ to } (\alpha, \beta \text{ or } \beta')</math></td><td><math>(\alpha, \beta \text{ or } \alpha, \beta') \text{ to } (\alpha_T, \phi)</math></td><td><math>(\alpha, \beta') \text{ to } (\alpha, \beta)</math></td></tr><tr><td><math>\tan \alpha = \tan \alpha_T \cos \phi</math></td><td><math>\cos \alpha_T = \cos \alpha \cos \beta</math></td><td><math>\tan \beta = \tan \beta' \cos \alpha</math></td></tr><tr><td><math>\sin \beta = \sin \alpha_T \sin \phi</math></td><td><math>\tan \phi = \tan \beta / \sin \alpha</math></td><td></td></tr><tr><td><math>\tan \beta' = \tan \alpha_T \sin \phi</math></td><td><math>\tan \alpha_T = \sqrt{\tan^2 \alpha + \tan^2 \beta'}</math></td><td></td></tr><tr><td></td><td><math>\tan \phi = \tan \beta' / \tan \alpha</math></td><td></td></tr></table>					$(\alpha_T, \phi) \text{ to } (\alpha, \beta \text{ or } \beta')$	$(\alpha, \beta \text{ or } \alpha, \beta') \text{ to } (\alpha_T, \phi)$	$(\alpha, \beta') \text{ to } (\alpha, \beta)$	$\tan \alpha = \tan \alpha_T \cos \phi$	$\cos \alpha_T = \cos \alpha \cos \beta$	$\tan \beta = \tan \beta' \cos \alpha$	$\sin \beta = \sin \alpha_T \sin \phi$	$\tan \phi = \tan \beta / \sin \alpha$		$\tan \beta' = \tan \alpha_T \sin \phi$	$\tan \alpha_T = \sqrt{\tan^2 \alpha + \tan^2 \beta'}$			$\tan \phi = \tan \beta' / \tan \alpha$	
$(\alpha_T, \phi) \text{ to } (\alpha, \beta \text{ or } \beta')$	$(\alpha, \beta \text{ or } \alpha, \beta') \text{ to } (\alpha_T, \phi)$	$(\alpha, \beta') \text{ to } (\alpha, \beta)$																	
$\tan \alpha = \tan \alpha_T \cos \phi$	$\cos \alpha_T = \cos \alpha \cos \beta$	$\tan \beta = \tan \beta' \cos \alpha$																	
$\sin \beta = \sin \alpha_T \sin \phi$	$\tan \phi = \tan \beta / \sin \alpha$																		
$\tan \beta' = \tan \alpha_T \sin \phi$	$\tan \alpha_T = \sqrt{\tan^2 \alpha + \tan^2 \beta'}$																		
	$\tan \phi = \tan \beta' / \tan \alpha$																		

Table 3. NAMELIST FLTCON Variables

### 3.1.2 Namelist REFQ - Reference Quantities

Inputs for this namelist are optional and are defined in **Table 4**. A vehicle scale factor (SCALE) permits the user to input a geometry that is scaled to the size desired. This scale factor is used as a multiplier to the user defined geometry inputs and to the user input reference quantities (SREF, LREF, LATREF, XCG). If no reference quantities are input, they are computed based upon the scaled geometry. XCG is input relative to the origin of the global coordinate system (X=0, **Figure 2**) and is scaled using SCALE.

NAMELIST REFQ				
VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
SREF	-	Reference area	L <sup>2</sup>	*
LREF	-	Longitudinal reference length	L	**
LATREF	-	Lateral reference length	L	LREF
XCG	-	Longitudinal position of C.G. (+aft)	L	0.
ZCG	-	Vertical position of C.G. (+up)	L	0.
BLAYER	-	Boundary layer type: TURB for fully turbulent NATURAL for natural transition	-	TURB
ROUGH ***	-	Surface roughness height (see <b>Table 5</b> for range of values)	L	0.
RHR ***	-	Roughness Height Rating (see <b>Table 6</b> for range of values)	-	0.
SCALE	-	Vehicle scale factor	-	1.
Note: * Default is maximum body cross-sectional area. If no body is input, default is maximum fin panel area. ** Default is maximum body diameter. If no body is input, default is fin panel mean geometric chord. *** Either ROUGH or RHR can be used. If ROUGH is used, the units must be inches (for English) or centimeters (for Metric).				

Table 4. NAMELIST REFQ Variables

In lieu of specifying the surface roughness height ROUGH, the surface Roughness Height Rating (RHR) can be specified. The RHR represents the arithmetic average roughness height variation in millionths of an inch. Typical values of ROUGH and RHR are given in **Table 6**.



Table 5. Equivalent Sand Roughness

TYPE OF SURFACE	EQUIVALENT SAND ROUGHNESS k (INCHES)	RHR
Aerodynamically Smooth	0.0	0.0
Polished Metal or Wood	0.00002 to 0.00008	6 to 26
Natural Sheet Metal	0.00016	53
Smooth Matte Paint, Carefully Applied	0.00025	83
Standard Camouflage Paint, Average Application	0.00040	133
Camouflage Paint, Mass Production Spray	0.0012	400
Dip Galvanized Metal Surface	0.006	2000
Natural Surface of Cast Iron	0.01	3333

Table 6. Preferred RHR Values

APPLICATION	RHR
Steel Structural Parts	250
Aluminum and Titanium Structural Parts	125
Close Tolerance Surfaces	63
Seals	32

### 3.1.3 Namelist AXIBOD - Axisymmetric Body Geometry

An axisymmetric body is defined using this namelist. The namelist input variables are given in **Table 7** and **Table 8** and a sketch of the geometric inputs are given in **Figure 2** and **Figure 3**. The body can be specified in one of two ways:

OPTION 1: The geometry is divided into nose, centerbody, and aft body sections. The shape, overall length, and base diameter for each section are specified. Note that not all three body sections need to exist on a configuration; for example, a nose-cylinder configuration does not require definition of an aft body.

**NOTE: Versions prior to the 1/06 release could NOT handle CONIC centerbodies. The 1/06 and subsequent releases include changes that now allow for conic centerbodies.**

OPTION 2: The longitudinal stations and corresponding body radii are defined, from nose to tail.

The program uses the input value for NX to determine which option is being used. If NX is not input then Option 1 inputs are assumed. If both shapes and body coordinates (Options 1 and 2) are used, the body coordinate information will take precedence.

If Option 2 is selected, the program generates a body contour based on the user specified values of X, R, and DISCON. Many additional points in between the user specified input coordinates will be generated. The resulting contour can contain more than 1000 points. If the PRINT GEOM BODY control card is used, this contour will be written to tape unit 9 ("for009.dat").

**NOTE: An option to specify a cambered body was added in the 1/06 and subsequent releases. This new functionality added the Z namelist input as an optional input for the Option 2 body input. Cambered should only be run with the aerodynamic roll angle (PHI) set to zero.**

It is highly recommended that Option 1 be used when possible. The program automatically calculates the body contour based upon the segment shapes using geometry generators. Hence, more accurate calculations are possible. **Even when Option 2 is used, appropriate Option 1 inputs should be included.** This identifies where the code should insert break points in the contour. If these parameters are not input, they are selected as defined in **Table 9**.

If DEXIT is not input, the base drag computed for the body geometry will not be included in the final computed axial force calculations. To include a "full" base drag increment, a zero exit diameter must be specified (DEXIT=0.).

If body coordinates are input using the variables NX, X, R, and DISCON, the nose is spherically blunted, and results using the Second Order Shock Expansion method are desired (only if  $M > 1.2$ ), the geometry must be additionally defined using the following:

- BNOSE must be specified
- TRUNC must be set to .FALSE.
- The first five (5) points in the X and R arrays must lie on the spherical nose cap with the fifth point being the transition from the spherical nose cap to the rest of the nose or body [i.e., X(1), X(2), X(3), X(4), X(5), R(1), R(2), R(3), R(4), and R(5) are spherical cap coordinates with X(5) and R(5) being the transition point].

The following summarizes the input generality available:

- X(1) does not have to be 0.0; an arbitrary origin can be selected.
- Five shapes can be specified by name:
 

CONICAL (CONE)	- cone or cone frustrum (default for boattails and flares)
OGIVE	- tangent ogive (default for noses)
POWER	- power law
HAACK	- L-V Haack (length-volume constrained)
KARMAN	- von Karman (L-D Haack; length-diameter constrained)
- If DAFT < DCENTR the afterbody is a boattail.
- If DAFT > DCENTR the afterbody is a flare.
- If LAFT is not input, aft body (boattail or flare) does not exist.

The inputs for base-jet plume interaction effects are defined using Option 1. Incremental forces and moments due to jet induced boattail separation and separation locations on aft fins are calculated if these inputs are used.

- This option should only be run for supersonic cases (i.e.  $M_\infty > 1.2$ )
- The calculations will be done for three types of aft bodies conical boattail, ogival boattail, or cylindrical (i.e. no boattail). Error messages will be printed to the output file and the calculations skipped if any other aft body is defined.
- If BASE=.FALSE. or is not input the calculations will be skipped.
- DEXIT must not equal zero if this option is used.
- The jet Mach number (JMACH), jet to freestream static pressure ratio (PRAT), and jet to freestream stagnation temperature ratio (TRAT) must be specified for each freestream Mach number or velocity input in the namelist FLTCON. For subsonic or transonic freestream Mach numbers or velocities, dummy values must be input for JMACH, PRAT, and TRAT. The user must be careful to match these inputs with the proper freestream conditions.
- If a portion of the fins in a fin set are located on the boattail or base, the boattail separation locations will be calculated and output at each fin roll

angle. However, if the fins do not extend to the boattail the separation locations will be skipped.

- Results may be inaccurate if excessive extrapolation is required. If extrapolation occurs, a warning message will be printed to the output file. To avoid extrapolation and minimize inaccuracy, the input parameters should be kept within the ranges shown in **Table 11**.

### ***NAMELIST AXIBOD - (Option 1 Inputs)***

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
TNOSE	-	Nose shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive) * POWER <or 2> (power law) HAACK <or 3> (L-V constrained) KARMAN <or 4> (L-D constrained)	-	OGIVE
POWER	-	Exponent, n, for power law shape: $(r/R)=(x/L)^n$	-	0.
LNOSE	-	Nose length	L	-
DNOSE	-	Nose diameter at base	L	1.
BNOSE	-	Nose bluntness radius or radius of truncation	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
LCENTR	-	Centerbody length	L	0.
DCENTR	-	Centerbody diameter at base	L	DNOSE
TAFT	-	Afterbody shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive)	-	CONICAL
LAFT	-	Afterbody length	L	0.
DAFT	-	Afterbody diameter at base (must be > 0 and not equal to DCENTR)	L	-
DEXIT		Nozzle diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. Gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-

Table 7. NAMELIST AXIBOD Variables (Option 1 Inputs)

### ***NAMelist AXIBOD – (Option 2 Inputs)***

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
NX	-	Number of input stations ( $2 < NX < 50$ )	-	-
X	50	Longitudinal coordinates X(NX) must be the end of the body	L	-
R	50	Radius at each X station	L	-
Z	50	Distance of bent missile centerline from the un-bent centerline.	L	0
DISCON	20	Indices of X stations where the surface slope is discontinuous. Example: X(1)=0.,4.,8.,12.,16.,20., DISCON=3., defines a discontinuity at X=8. (third value)	-	-
BNOSE	-	Nose bluntness radius or radius of truncation	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
DEXIT	-	Nozzle diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. gives “full” base drag DEXIT= exit gives base drag of annulus around exit only	L	-
<p>Note: LNOSE, DNOSE, LCENTR, DCENTR, LAFT, DAFT, and DEXIT should also be defined when using the AXIBOD option 2 input scheme. If they are not specified, Missile Datcom will use the default values defined in <b>Table 9</b>.</p>				

Table 8. NAMelist AXIBOD Variables (Option 2 Inputs)

LNOSE	Length of the body segment to where the radius first reaches a maximum
DNOSE	The diameter at the first radius maximum
LCENTR	Length of the body segment
DCENTR	Diameter at the end of the body segment
LAFT	The second body segment
DAFT	Diameter at the base of the second body segment
DEXIT	Not defined (implies that base drag is not to be included in the axial force calculations)

Table 9. Default values assigned when using AXIBOD Option 2 input scheme.

***NAMelist AXIBOD (Base-Jet Plume Interaction Inputs)***

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
BASE *	-	Flag for base plume interaction: .TRUE. for plume calculations .FALSE. for no plume calculations	-	.FALSE.
BETAN *	-	Nozzle exit angle	deg	-
JMACH **	20**	Jet Mach number at nozzle exit	-	-
PRAT **	20**	Jet/freestream static pressure ratio	-	-
TRAT **	20**	Jet/freestream stagnation temperature ratio	-	-
Note: * Only required if base plume interaction calculations are desired. ** JMACH, PRAT and TRAT must be specified for each freestream Mach number or velocity input in Namelist \$FLTCON.				

Table 10. NAMelist AXIBOD Base-Jet Plume Interaction Variables

Input Parameter	Symbol	Min Value	Max Value
Boattail shape	--	Cylinder, Cone, Ogive	
Boattail fineness ratio	$L/D$	0	2
Boattail terminal angle	$\beta_E$	0	12
Jet pressure ratio	$P_J/P_{inf}$	0	10
Freestream Mach number	$M$	2	5
Angle of Attack	$\alpha$	0	8
Jet Mach number	$M_J$	$M_{inf} - 1$	$M_{inf} + 1$
Nozzle terminal angle	$\beta_N$	5	25
Jet diameter ratio	$D_J / D_B$	0.80	0.95
Jet temperature ratio	$T_J / T_{inf}$	4	10

Note: If input parameter is not between minimum and maximum values the code will extrapolate.

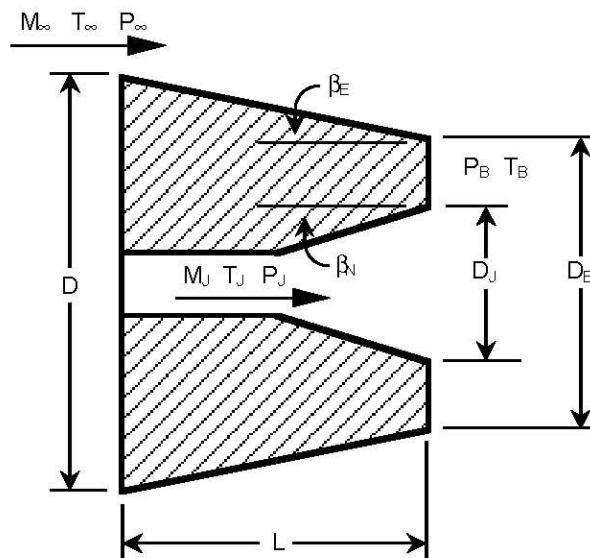


Table 11. Base-Jet Plume Interaction Parameter Limitations



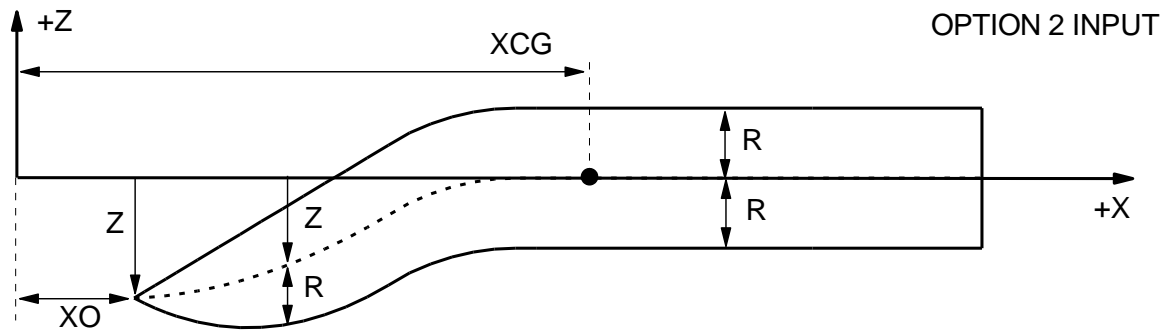
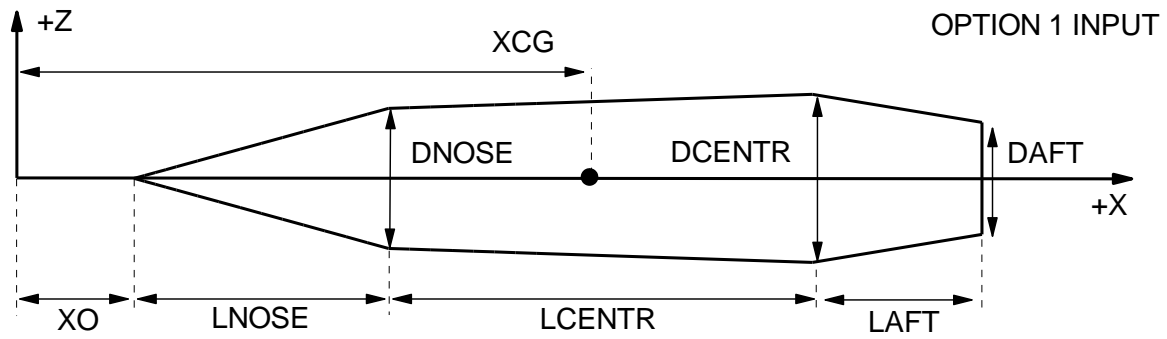


Figure 2. Axisymmetric Body Geometry Variables

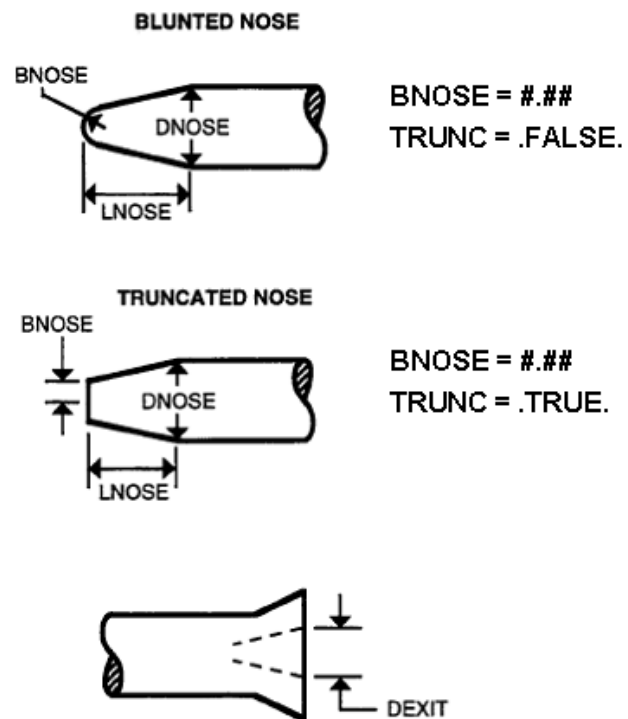


Figure 3. Blunted and Truncated Nose Variables

#### 3.1.4 Namelist ELLBOD - Elliptical Body Geometry

Elliptically-shaped cross section bodies are defined using this namelist. The inputs are similar to those for the axisymmetric body geometry (AXIBOD), and are shown in **Table 12** and **Table 13**. The types of shapes available, and the limitations, are the same as those given for axisymmetric bodies. **However, the base-jet plume interaction input options in namelist AXIBOD are not available in namelist ELLBOD.** Please read **Section 3.1.3** for limitations.

Note that the body cross section ellipticity can vary along the body longitudinal axis. Sections which are taller-than-wide and wider-than-tall can be mixed to produce "shaped" designs. The shape of the sections is controlled by the variables ENOSE, ECENTR, and EAFT or ELLIP, H and W.

### ***NAMELIST ELLBOD – (OPTION 1 INPUTS)***

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
TNOSE	-	Nose shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive)* POWER <or 2> (power law) HAACK <or 3> (L-V constrained) KARMAN <or 4> (L-D constrained)	-	OGIVE
POWER	-	Exponent, n, for power law shape: $(r/R)=(x/L)^n$	-	0.
LNOSE	-	Nose length	L	-
WNOSE	-	Nose width at base	L	1.
ENOSE	-	Ellipticity at nose base (height/width)	-	1.0
BNOSE	-	Nose bluntness radius or radius of truncated nose. <b>BNOSE is relative to width (WNOSE). Height is calculated by BNOSE * ENOSE</b>	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
LCENTR	-	Centerbody length	L	0.
WCENTR	-	Centerbody width at base	L	WNOSE
ECENTR	-	Ellipticity at centerbody base (height/width)	-	1.0
TAFT	-	Afterbody shape and numerical equivalent: CONICAL or CONE <or 0> (cone) OGIVE <or 1> (tangent ogive)	-	CONICAL
LAFT	-	Afterbody length	L	0.
WAFT	-	Afterbody diameter at base (must be > 0 and not equal to WCENTR)	L	-
EAFT	-	Ellipticity at aft body base (height/width)	-	1.0
DEXIT	-	Nozzle equivalent diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-

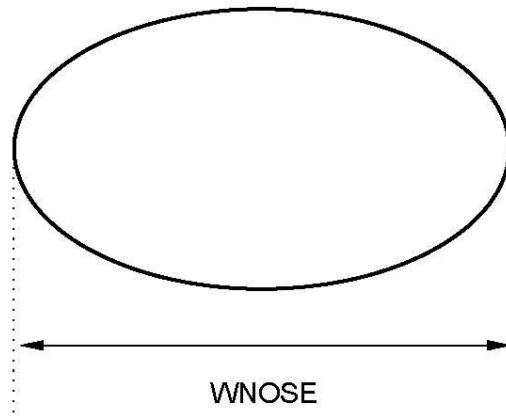
Table 12. NAMELIST ELLBOD Variables (Option 1 Inputs)

### ***NAMelist ELLBOD – (Option 2 Inputs)***

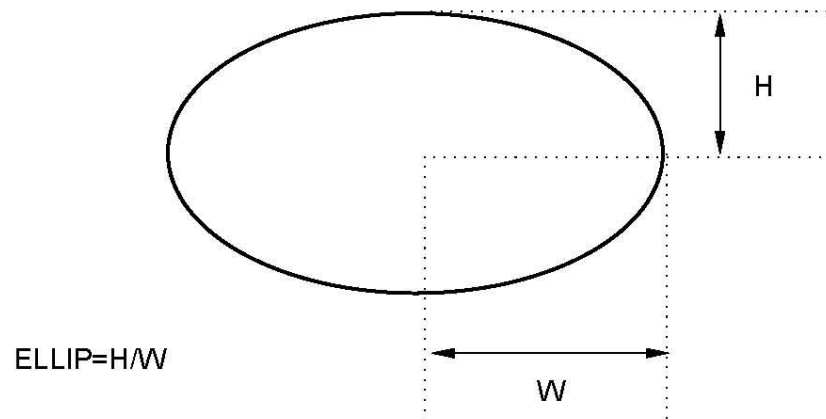
<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
XO or X0	-	Longitudinal coordinate of nose tip	L	0.
NX	-	Number of input stations ( $2 < NX < 50$ )	-	-
X	50	Longitudinal coordinates X(NX) must be the end of the body	L	-
H *	50	Body half-height at each X station		
W *	50	Body half-width at each X station	L	-
ELLIP *	50	Body height to width ratio at each X station	-	1.0
DISCON	20	Indices of X stations where the surface slope is discontinuous. Example: X(1) = 0., 4., 8., 12., 16., 20., DISCON=3., defines a discontinuity at X=8. (third value)	-	-
BNOSE	-	Nose bluntness radius or radius of truncated nose. <b>BNOSE is relative to width (WNOSE). Height is calculated by BNOSE * ENOSE</b>	L	0.
TRUNC	-	Truncation flag: .TRUE. is nose is truncated .FALSE. is nose is not truncated	-	.FALSE.
DEXIT	-	Nozzle equivalent diameter for base drag calculation DEXIT not defined gives zero base drag DEXIT = 0. gives "full" base drag DEXIT= exit gives base drag of annulus around exit only	L	-
Note: * One of the following combinations is required: 1. W and H 2. W and ELLIP 3. H and ELLIP				

Table 13. NAMelist ELLBOD Variables (Option 2 Inputs)

OPTION 1



OPTION 2



NOTE: Option 1 input  $W_{NOSE}$  is TOTAL Width, Option 2 input  $W$  is HALF width

Figure 4. Elliptical Body Variables

### 3.1.5 Namelist PROTUB - Protuberance Geometry

Missile protuberances can be input using this namelist. Axial force coefficient is calculated for the protuberances and added to the body axial force coefficient. **Table 14** shows the inputs required. **Figure 5** shows the different protuberance shapes available. The following defines the inputs required for protuberance calculations:

- NPROT is the number of protuberance sets. A protuberance set is made up of protuberances at the same axial location with the same size and shape. Therefore, it is only necessary to describe the geometry of one individual protuberance per set. The maximum number of protuberance sets is 20.
- NLOC is the number of protuberances in each protuberance set. NLOC accounts for the number of identical protuberances located around the missile body at a given axial location.
- The following equation helps to clarify the relationship between NLOC and NPROT:  
$$NLOC(1)+NLOC(2)+NLOC(3)+ \dots +NLOC(NPROT) = (\text{Total number of protuberances on the missile})$$
- The axial location of a protuberance (XPROT) should be input at the protuberance geometric centroid. An approximation of the centroid will be adequate for the analysis. The location is used to calculate the average boundary layer thickness over the protuberance length.

**NOTE: The angular orientation of protuberances can be defined with PHIPRO. This variable is only used to calculate a moment from the axial force of the protuberance. If PHIPRO is not defined, no moment will be computed.**

- VCYL, HCYL, BLOCK, and FAIRING type protuberances have 1 member. LUG types have 4 members and SHOE types have 3 members (**Figure 5**).
- All inputs for LPROT, WPROT, HPROT, and OPROT are in sequential order based upon the members specified with the protuberance type (PTYPE) input.
- The FAIRING type protuberance should always have a zero offset. The code will assume a zero offset even if a non-zero offset is input.

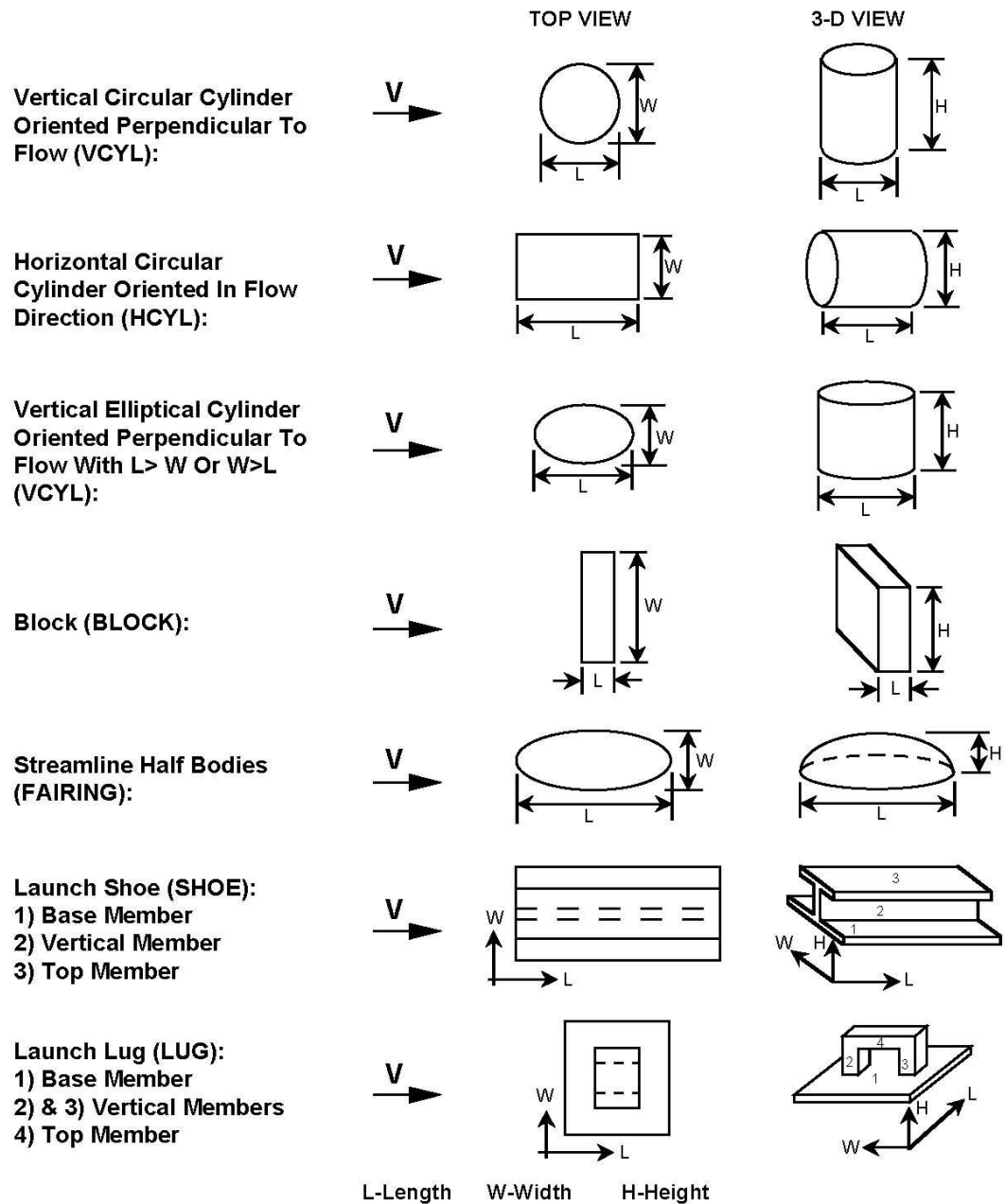
More complex protuberance shapes can be analyzed by a component build-up method. Each member is treated as a separate protuberance. Combinations of vertical cylinders, horizontal cylinders, and flat plates or blocks can be input at specified offsets from the missile body. If a FAIRING type protuberance is used in a component build-up, the offset should be zero. The user must manually add axial force of the individual members of the component build-up if the total protuberance axial force is desired.

**Figure 6** shows an example input file for a missile with several protuberances.

### ***NAMelist PROTUB***

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
NPROT	-	Number of protuberance sets (20 maximum)	-	0.
PTYPE	20	Protuberance set type: VCYL <or 1.> (cyl. perp. to flow) HCYL <or 2.> (cyl. aligned with flow) LUG <or 3.> (launch lug) ** SHOE <or 4.> (launch shoe)** BLOCK <or 5.> FAIRING <or 6.> (streamline half body)	-	-
XPROT	20	Longitudinal distance from missile nose to the geometric centroid of the protuberance set	L	-
NLOC *	20	Number of protuberances in set	-	0.
PHIPRO***	400	Protuberance angular orientation measured clockwise from top vertical center looking forward	deg	-
LPROT	100	Length of protuberance	L	-
WPROT	100	Width of protuberance	L	-
HPROT	100	Height of protuberance	L	-
OPROT	100	Vertical offset of protuberance	L	0.
<b>Note:</b> * NLOC defines for identical protuberances (same size and shape) located around the body at the same axial station. ** LUG type has 4 members. SHOE type has three members. LPROT, WPROT, HPROT, and OPROT must be specified for each member. *** Angles for each protuberance are entered in the same order as the sets. An angle must be defined for each protuberance if this option is used. Example: A case has 2 protuberance sets with 3 protuberances in the first set (0, 120, and 240 degrees) and 2 in the second set (90, 270). Enter the data as NPROT = 2, NLOC = 3, 2, PHIPRO = 0, 120, 240, 90, 270,				

Table 14. NAMelist PROTUB Variables

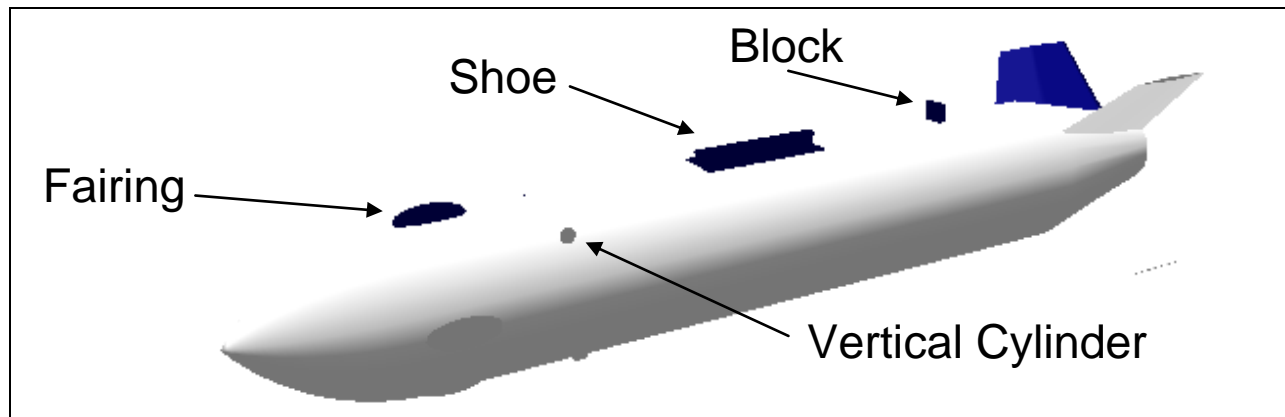


Note; Length, width, height, and offset must be input for each member of launch lug and launch shoe types

Note; Offset is the perpendicular distance from the missile mold line to the bottom of the protuberance or protuberance member

Figure 5. Available Protuberances Shapes





```

CASEID Protuberance Example Case
DIM IN
$FLTCON
  NALPHA = 4.00,
  ALPHA = 0.00, 2.00, 4.00, 6.00,
  NMACH = 3.00,
  MACH = 0.40, 0.80, 2.00,
  REN = 3.0000E+06, 3.0000E+06, 3.0000E+06,
$END
$REFQ
  SREF = 113.0973,
  LREF = 12.,
  XCG = 39.,
  BLAYER = TURB,
$END
$AXIBOD
  TNOSE = OGIVE,
  LNOSE = 12.0000,
  DNOSE = 12.0000,
  TRUNC = .FALSE.,
  LCENTR = 54.0000,
  DCENTR = 12.0000,
  TAFT = CONICAL,
  LAFT = 12.0000,
  DAFT = 6.0000,
$END
$PROTUB
  NPROT = 4.,
  PTYPE = FAIRING, VCYL, SHOE, BLOCK,
  XPROT = 14., 22., 39., 56.,
  NLOC = 4., 4., 1., 1.,
  LPROT = 5.00, 1.00, 10.00, 10.00, 10.00, 0.50,
  WPROT = 2.00, 1.00, 4.00, 0.25, 1.00, 2.00,
  HPROT = 1.00, 0.50, 0.10, 0.75, 0.25, 1.50,
  OPROT = 0.00, 0.00, 0.00, 0., 0., 0.00,
  PHIPRO = 0., 90., 180., 270., 45., 135., 225., 315., 0., 0.,
$END
$FINSET1
  SECTYP = HEX,
  SSPAN = 0.0000, 9.0000,
  CHORD = 14.0000, 8.0000,
  XLE = 64.0000,
  SWEEP = 0.0000,
  STA = 1.0000,
  NPANEL = 4.0000,
  PHIF = 45.0000, 135.0000, 225.0000, 315.0000,
$END
SAVE
NEXT CASE

```

Figure 6. Sample Missile with Protuberances

### 3.1.6 Namelist FINSETn - Define Fin Set n

**Table 15** describes the variables needed to be input for fin set planform geometry descriptions. Optional fin cross-section inputs are described in **Figure 8**. Special user specified fin cross-sections can be input using the variables in **Figure 9**. **The user may specify up to nine sets of fins.** The variable "n" in the namelist specifies the fin set number. Fin sets are normally numbered sequentially from the front to the back of the missile beginning with fin set one. An input error will occur if "n" is zero or omitted. The code allows for between 1 and 8 geometrically identical panels to be input per fin set. The panels may be arbitrarily rolled about the body and can be given dihedral.

Four types of airfoil sections are permitted--hexagonal (HEX), circular arc (ARC), NACA airfoils (NACA), and user defined (USER). Only one type of airfoil section can be specified per fin set, and this type is used for all chord wise cross sections from root to tip. Diamond-shaped sections are considered a special case of the HEX type; hence, hexagonal and diamond sections can coexist on the same panel. The airfoil proportions can be varied from span station to span station. **Table 16** describes the input variables for HEX and ARC airfoil sections. **Table 17** described the input variables for USER airfoil sections. It is not possible in the current version to specify a cambered airfoil using the HEX or ARC airfoil definitions. Cambered airfoils can only be defined with a USER or NACA airfoil definition.

**NOTE: HEX and ARC airfoils can vary across the span, and must be defined at each span station for each fin set where they are used. If they are only defined once (at the root chord), the default values will be used at all other chord stations. NACA and USER airfoils cannot vary across the span and are only defined once for each fin set.**

The user selects "break points" on the panel (**Figure 7**). A "break point" specifies a change in leading or trailing edge sweep angle. Also a break point may specify a change in airfoil section, but the section must be of the same type (i.e., a change in section type cannot go from a NACA to an ARC) only the proportions can change. The location of each "break point" is defined by specifying its semi-span station (SSPAN) from the vehicle centerline and distance from the first body station to the chord leading edge (XLE). The "break point" chord leading edge array (XLE) can be defined by simply specifying the root chord leading edge [XLE(1)] and the sweep angles of each successive panel segment if the semi-span stations are input. Note that only those variables that uniquely define the fin need to be entered. Redundant inputs can lead to numerical inconsistencies and subsequent computational errors.

The panel sweep angle (SWEEP) can be specified at any span station for each segment of the panels. If STA=0., the sweep angle input is measured at the segment leading edge; if STA=1., the sweep angle input is measured at the segment trailing edge. Note that some aerodynamic methods are very sensitive to panel sweep angle. For small span fins, small errors in the planform inputs can create large sweep angle calculation errors. It is recommended that exact sweep angles be specified wherever possible; for example, if the panel trailing edge is unswept, specifying SWEEP=0. and STA=1. will minimize calculation error. Then the leading edge sweep will be computed by the code internally using the SSPAN and CHORD inputs.

**SSPAN(1):**

It is the user's responsibility to assure that the fins are (1) on the body surface, and (2) do not lie internal to the body mold line. The program does not check for these peculiarities. If SSPAN(1)=0. is input, the program will assume that the panel semi-span data relative to its root chord are supplied. The code will automatically interpolate the body geometry to place the panel on the body surface with the root chord parallel to the body centerline, so that the center of the root chord is on the surface mold line (see **Figure 7**).

**CFOC:**

Plain trailing edge devices may be modelled in Missile Datcom. This is accomplished via the CFOC array which is the flap chord to fin chord ratio,  $c_f/c$ . Trailing edge devices can be either full span or partial span subject to certain limits specified below. The trailing edge devices can not have a taper ratio greater than 1.0, and the hinge line must be straight regardless of the number of segments comprising the trailing edge device. A partial span trailing edge device is specified by setting CFOC=0 for those chord/span stations that are not part of the trailing edge device. Examples of acceptable and unacceptable geometries are shown in **Figure 8** as well as the corresponding input values for the variable arrays CFOC, CHORD and SSPAN. **A special case where the trailing edge device extends to the tip of a fin with a taper ratio of zero** is also shown in **Figure 8**. While any value of CFOC will result in the correct flap chord at the tip (since the tip chord is zero), the user must specify a non-zero CFOC since a value of CFOC=0 would indicate the trailing edge device does not exist at this chord/span station. The user should also be aware of the following:

- All trailing edge deflection angles are measured with respect to the freestream and not relative to the hinge line. This becomes an important distinction as the hinge line sweep angle is increased. The following equation converts the hinge line sweep to the freestream sweep:  $\delta_{flap} = \tan^{-1}(\tan \delta_{HingeLine} \cos \Lambda_{HingeLine})$
- The variable SKEW does not apply to trailing edge devices.
- The hinge moments for trailing edge devices are not calculated.
- The increase in profile drag due to trailing edge deflection is not calculated.

Internally, Missile Datcom calculates the flap chord ratio between non-zero values of CFOC. For example,

$$CFOC = 0.0, 0.30, 0.25, 0.0,$$

This fin panel has three sections. The inner most section has no flap as CFOC(1) is set to zero, The center panel has a flap that varies from 30% of the chord length at the spanwise beginning of the panel and varies linearly to 25% of the chord length at the end of the center panel. The end most panel has no flap as CFOC(4) is set to zero.

**NPANEL, PHIF and GAM:**

Since all panels are assumed to be planar (i.e., no tip dihedral), all inputs must be "true view". Once the planform of a single panel is defined, all fins of the set are assumed to be identical. The number of panels present is defined using the variable NPANEL. Each panel may be rolled to an arbitrary position around the body using the variable PHIF. PHIF is measured clockwise from top vertical center (looking forward from behind the missile) as shown in **Figure 9**. Each panel may also contain a constant dihedral. A panel has zero dihedral when it is aligned along a radial ray from the centerline (see **Figure 9**). For a circular body, this makes the fin perpendicular to the body mold line. For an elliptical body, the fin may not be

perpendicular to the body mold line if there is no dihedral. The variable used to specify dihedral is GAM. GAM is positive if the panel tip chord is rotated clockwise.

Different aerodynamics will be computed depending upon whether the FLTCOIN namelist variable PHI, or the FINSETn namelist variable PHIF, is used to roll the geometry. **Figure 10.** depicts the usage of the roll options. The variable "PHI" means that the body axes system is to be rolled with the missile body, whereas PHIF keeps the aerodynamics in a non-rolled body axis, but rather locates the fin positions around the body. PHIF must be input for each panel, while PHI rolls the whole configuration.

**NOTE: When defining more than one fin set, the sets should be input in order as they are mounted on the body from nose to tail. This will ensure correct tracking of the fin vortices. The code only tracks vortices from a given finset to a finset with a higher index. For example, vortices from FINSET2 will be tracked to FINSET3 and FINSET4 only, not to FINSET1.**

**NOTE: The code will not track vortices between finsets if there is not a positive longitudinal gap between the trailing edge of the forward fin and leading edge of the aft fin. This allows multiple fin planforms to be defined at the same longitudinal station, for example, a horizontal and vertical tail of differing planform.**

There should be sufficient space between the forward fin trailing edge and aft fin leading edge to avoid violating the assumptions made by the aerodynamic computations. It is assumed by the aerodynamic model that the vortices are fully rolled up when they pass the control points of the next downstream set of fins. In reality the vortex sheet does not fully roll up until it is at several semispans downstream. If two fin sets are closer than this the results may be in error since the use of a vortex filament model may introduce too much vorticity. The closer the spacing the larger the error may be. No algorithm error will result from too close a fin set spacing.

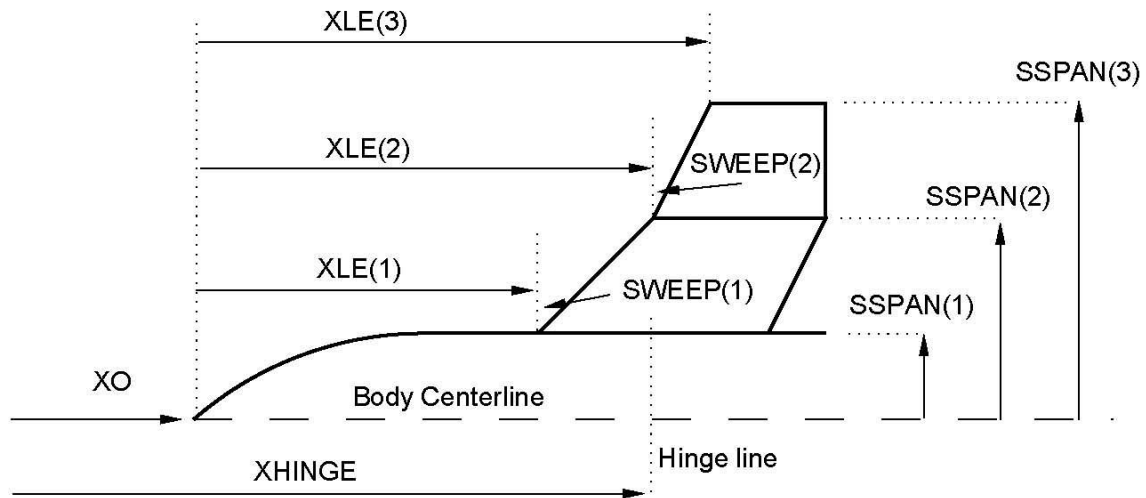
Panels with cut-out portions can be modeled by using one of the ten available fin segments as a transition segment. This is accomplished by giving the segment a small span, such as 0.0001, and specifying the segment root and tip chords to transition into the cut-out portion of the fin.

### ***NAMELIST FINSETn – (NOMINAL INPUTS)***

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
SECTYP	-	Airfoil section type and numerical equivalent: HEX <or 0.> (Hexagonal section) NACA <or 1.> (Requires NACA control card) ARC <or 2.> (Circular arc section) USER <or 3.> (User defined airfoil section)	-	HEX
SSPAN	10	Semi-span locations. To automatically place fin on body moldline, use SSPAN(1)=0.0 with other values relative to fin root chord.	L	-
CHORD	10	Panel chord at each semi-span location	L	-
XLE	10	Distance from missile nose to chord leading edge at each span location. Specify only XLE(1) if using SWEEP to define planform.	L	0.0
SWEEP	9	Sweepback angle at each span station.	deg	0.0
STA	9	Chord station used in measuring sweep: STA=0.0 is leading edge STA=1.0 is trailing edge	-	1.0
LER	10	Leading edge radius at each span station. Not required if SECTYP=NACA	L	0.0
NPANEL	8	Number of panels in fin set (1-8)	-	4
PHIF *	8	Roll angle of each fin measured clockwise from top vertical center looking forward	deg	*
GAM	8	Dihedral of each fin, positive when PHIF is Increased, see <b>Figure 9</b> .	deg	0.0
CFOC	10	Flap chord to fin chord ratio at each span station	-	1.0
Note: * If PHIF not used, panels will be evenly spaced around the body with the first fin placed at PHIF = 0.				

Table 15. NAMELIST FINSETn Variables – (Nominal Inputs)

### Multi-Segment Fin Placement



NOTE 1: XLE measured from body nose, XHINGE measured from origin

NOTE 2: Define either XLE(1) with various values of SWEEP  
OR multiple values of XLE with no SWEEP

### Varying Body Radius Placement

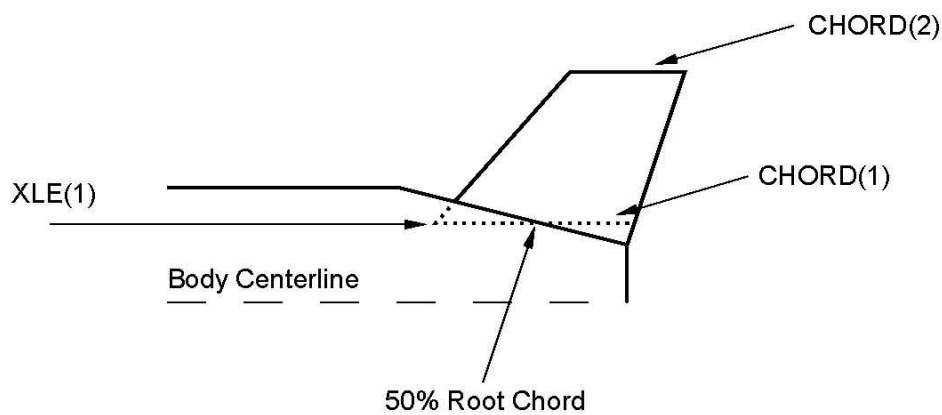
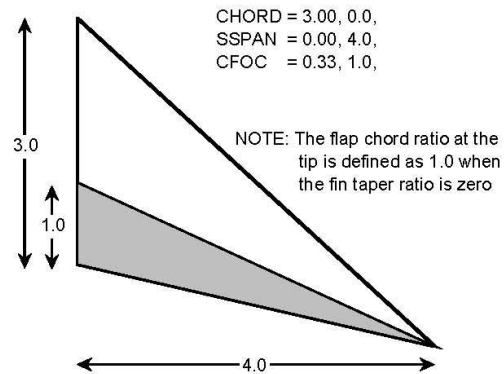
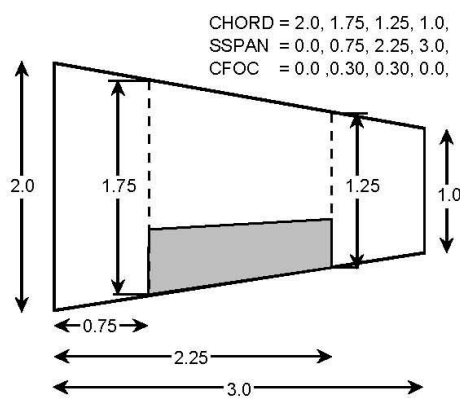
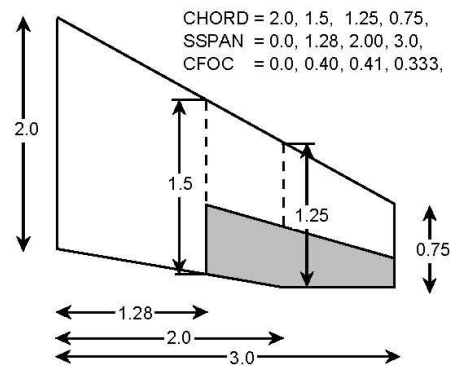
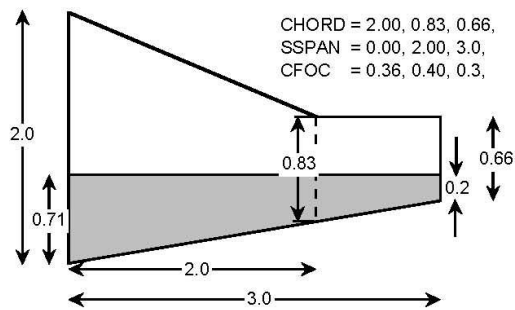
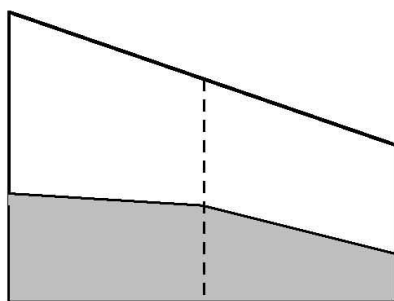


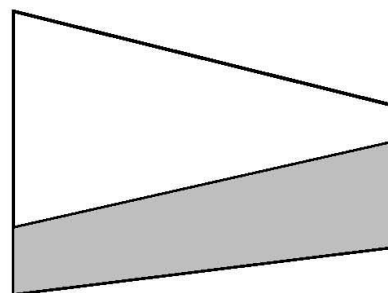
Figure 7. Fin Placement on Body



### Valid Trailing Edge Device Definitions



- Cranked Hinge Line Is Invalid



- Trailing Edge Device With Taper Ratio Greater Than One Is Invalid

### Invalid Trailing Edge Device Definitions

Figure 8. Definition of Trailing Edge Flaps

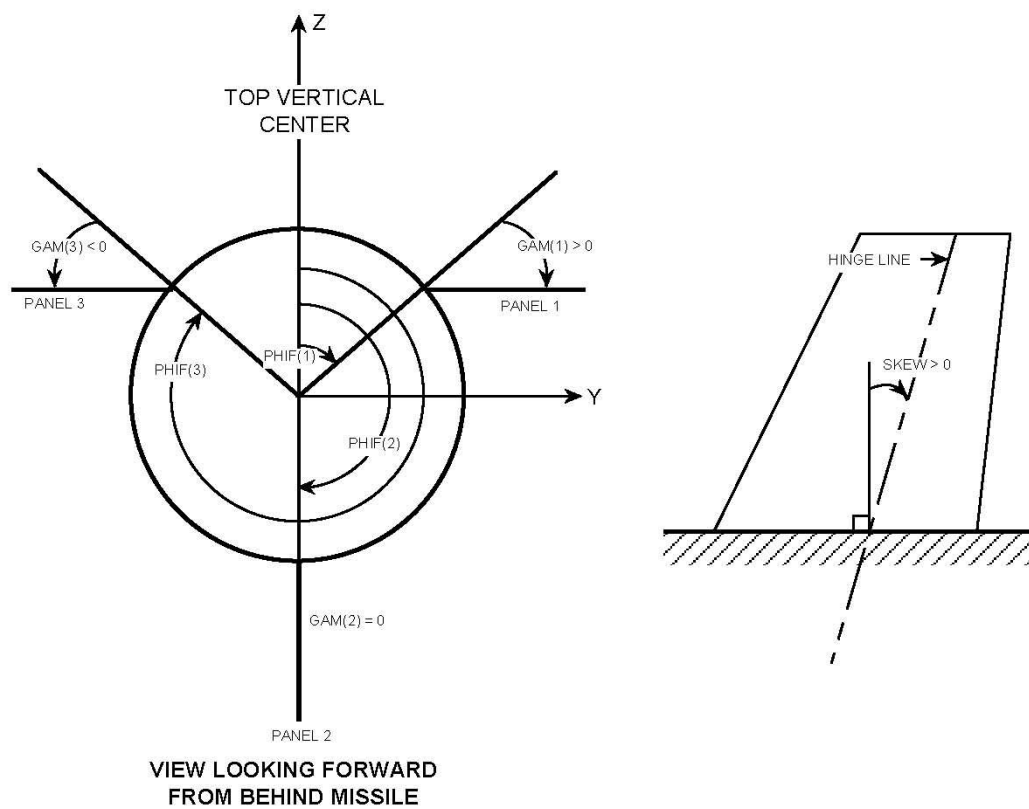
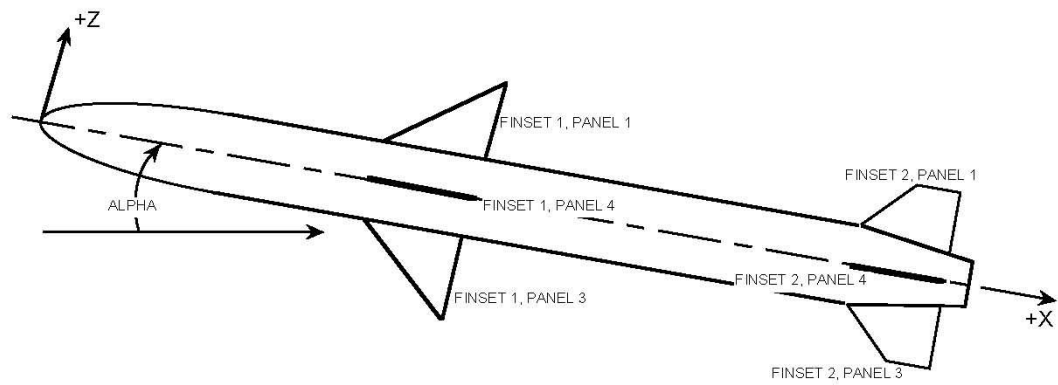
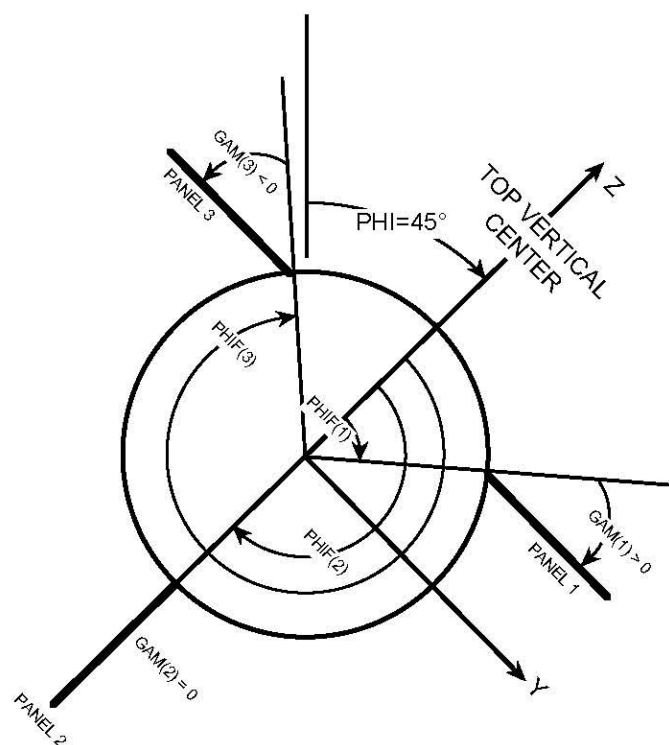


Figure 9. Fin Numbering and Orientation





VIEW LOOKING FORWARD  
FROM BEHIND MISSILE

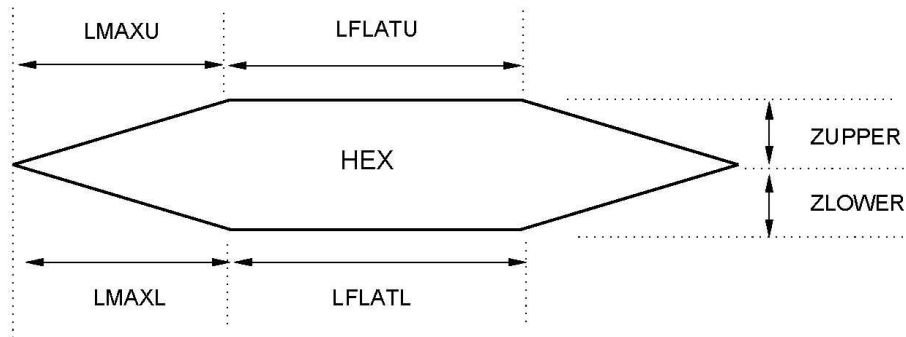
PHI IS THE BODY ROLL ANGLE  
PHIF IS THE FIN PANEL ROLL ANGLE

Figure 10. Roll Attitude vs Fin Orientation

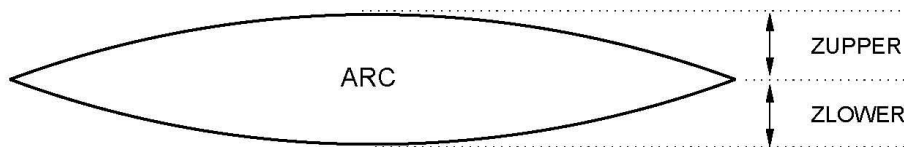
### NAMELIST FINSETn - (SECTYP= HEX, ARC INPUTS)

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
ZUPPER	10	Thickness to chord ratio of upper surface. Input separate value for each span station.	-	0.025
ZLOWER	10	Thickness to chord ratio of lower surface. Input separate value for each span station.	-	ZUPPER
LMAXU	10	Fraction of chord from leading edge to maximum thickness of upper surface. Input separate value for each span station.	-	0.5
LMAXL	10	Fraction of chord from leading edge to maximum thickness of lower surface. Input separate value for each span station.	-	LMAXU
LFLATU*	10	Fraction of chord of constant thickness section of upper surface. Input separate value for each span station.	-	0.
LFLATL*	10	Fraction of chord of constant thickness section of lower surface. Input separate value for each span station.	-	LFLATU
NOTE: All values must be defined at each span station				
* The variables LMAXL or LFLATL are not used in any aerodynamic calculation. They remain in the code for consistency and possible future use.				

Table 16. NAMELIST FINSETn Variables- (SECTYP= HEX, ARC inputs)



NOTE: All parameters must be input at each span station



NOTE: ARC section only allows ZUPPER and ZLOWER

Figure 11. HEX and ARC Airfoil Variables

### ***NAMelist FINSETn - (SECTYP= USER INPUTS)***

VARIABLE NAME*	ARRAY SIZE **	DEFINITION	UNITS	DEFAULT
XCORD	50	Chord station, fraction of chord from leading edge. Values must range from 0.0 (leading edge) to 1.0 (trailing edge).	-	-
MEAN	50	Distance between the mean line and chord at each XCORD station in fraction of chord	-	-
THICK	50	Thickness to chord ratio at each XCORD station	-	-
YUPPER ***	50	Upper surface coordinates at each XCORD station in fraction of chord	-	-
YLOWER ***	50	Lower surface coordinates at each XCORD station in fraction of chord	-	-
<p>Note:</p> <p>* One of the following combinations is required:</p> <ol style="list-style-type: none"> <li>1. XCORD, MEAN and THICK</li> <li>2. XCORD, YUPPER and YLOWER</li> </ol> <p>** Values apply to the entire fin set</p> <p>*** The airfoil must be closed at the leading and trailing edges, i.e. YUPPER=YLOWER or THICK=0.</p>				

Table 17. NAMelist FINSETn - (SECTYP= USER Inputs)

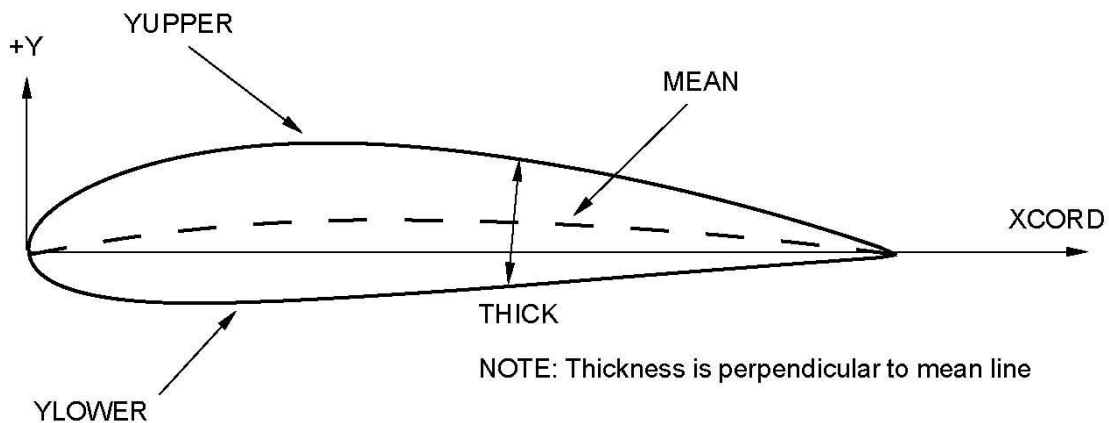


Figure 12. USER Airfoil Variables

Table 18. Airfoil Designation Using the NACA Control Card

NACA Series	Digit	Series Designation Rules	Samples (fin set and series follow NACA)
4 digit Series	1 2 3, 4	Maximum value of mean line ordinate (% chord) Distance to maximum thickness (tenths of chord) Maximum Thickness (% chord)	NACA-1-4-0008 NACA-2-4-2412
Modified 4 digit Series	1 – 4 5 6 7	Same as 4 digit series 'dash' (i.e. '-') Leading Edge Radius: 0 = sharp, 6 = normal radius Position of maximum thickness (tenths of chord), must be 2, 3, 4, 5, or 6	NACA-1-4-0012-62 NACA-2-4-4410-35 NACA-3-4-2004-04
5 digit Series	1 2, 3 4, 5	2/3 of design lift coefficient in tenths (2 indicates design Cl of 0.3) Twice distance to maximum camber point (% chord, i.e. 20 yields maximum camber point at 10% chord) Maximum thickness (% chord)	NACA-1-5-23012 NACA-2-5-42008
Modified 5 digit Series	1 – 5 6 7 8	Same as 5 digit series 'dash' (i.e. '-') Leading Edge Radius: 0 = sharp, 6 = normal radius Position of maximum thickness (tenths of chord), must be 2, 3, 4, 5, or 6	NACA-1-5-23012-32 NACA-2-5-42008-05 NACA-3-5-22406-63
1 Series	1 2 3 4 5, 6	Series designation Distance to minimum pressure point (tenths of chord, must be 6, 8, or 9) 'dash' (i.e. '-') Design lift coefficient in tenths Maximum thickness (% chord)	NACA-1-1-16-212 NACA-2-1-18-006 NACA-3-1-19-110
6 series	1 2 3 4 5, 6 7	Series designation Distance to minimum pressure point (tenths of chord) 'dash' (i.e. '-'): conventional section 'A': section straight from 80% chord to TE Design lift coefficient in tenths Maximum thickness (% chord) Optional mean line parameter (A=xx), must be decimal between 0.1 and 1.0 (default = 1.0)	NACA-1-6-64-005 NACA-2-6-61-205 A=0.6 NACA-3-6-65A010
Supersonic – Diamond or Circular Arc	1 2 3, 4 5 6	Type: 1 for diamond, 2 for circular arc 'dash' (i.e. '-') Distance to Maximum thickness (% chord) 'dash' (i.e. '-') Maximum thickness (% chord)	NACA-1-S-1-45-6 NACA-1-S-2-50-7
Supersonic – Hexagonal	1 2 3,4 4 5 6 7	Type: 3 'dash' (i.e. '-') Distance to Maximum thickness (% chord) 'dash' (i.e. '-') Maximum thickness (% chord) 'dash' (i.e. '-') Length of constant thickness section (% chord)	NACA-1-S-3-25-5-50
Note: Thickness can be expressed to nearest 0.01% chord for 1,4,5 and 6 series sections and nearest 0.1% chord for supersonic series. Examples: NACA-1-4-4411.75, NACA-1-S-1-50-7.5			

NOTE: A cambered section is oriented to provide positive normal force (body axes) at all angular orientations. For top and bottom fins ( $\phi=0$  or  $180$ ), the camber is oriented to provide positive side force.

### 3.1.7 Namelist DEFLCT - Panel Deflection Angles

This namelist permits the user to fix the incidence angle for each panel in each fin set. The variables are given in **Table 19**. Note that the panel numbering scheme is assumed to be that shown in **Figure 9**. The array element of each deflection array corresponds to the panel number.

The scheme for specifying deflection angles is unique, yet concise. The scheme used is based upon the body axis rolling moment:

*"In Missile Datcom a positive panel deflection is one which will produce a negative (counterclockwise when viewed from the rear) roll moment increment at zero angle of attack and sideslip."*

#### **NAMelist DEFLCT**

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
DELTA1	8	Deflection angles for each panel in fin set 1	deg	0.0
DELTA2	8	Deflection angles for each panel in fin set 2	deg	0.
DELTA3	8	Deflection angles for each panel in fin set 3	deg	0.
DELTA4	8	Deflection angles for each panel in fin set 4	deg	0.
DELTA5	8	Deflection angles for each panel in fin set 5	deg	0.
DELTA6	8	Deflection angles for each panel in fin set 6	deg	0.
DELTA7	8	Deflection angles for each panel in fin set 7	deg	0.
DELTA8	8	Deflection angles for each panel in fin set 8	deg	0.
DELTA9	8	Deflection angles for each panel in fin set 9	deg	0.
XHINGE *	4	Position of the panel hinge line for each fin set, measured from the coordinate system origin. XHINGE is NOT measured from the body nose unless XO=0.	L	XO+XLE+ CR/2*
SKEW	4	Hinge line sweepback for each fin set	deg	0.
Note: * Default is at one-half of the exposed root chord, as measured from the coordinate system origin. <b>A POSITIVE DEFLECTION ANGLE PRODUCES A NEGATIVE BODY AXIS ROLLING MOMENT AT ZERO ANGLE OF ATTACK.</b>				

Table 19. NAMelist DEFLCT Variables

### 3.1.8 Namelist TRIM - Trim Aerodynamics

This namelist instructs the program to statically trim the vehicle longitudinally ( $C_M=0$ ). The inputs are given in **Table 20**. Note that only one fin set can be used for trimming. The user only specifies the range of deflection angles desired using DELMIN and DELMAX; the code will try to trim the vehicle for each angle of attack specified using the allowable fin deflections. This option will not trim the vehicle at a specific angle of attack if the deflection required is outside the range set by the values of DELMIN and DELMAX.

The deflection sign convention used is that described in **Section 3.1.7**; hence, DELMIN and DELMAX are input as if deflecting the panel to the maximum will produce a negative rolling moment from the panels resulting normal force increment. DELMIN must always be less than DELMAX.

**NOTE: The logical variable ASYM has been deleted from the 8/08 release of the code.**

Missile Datcom takes DELMIN and DELMAX and divides the increment into nine equal segments, resulting in ten different deflection cases that are analyzed. From this data, Missile Datcom locates the point where the SIGN on CM changes and then linearly interpolates to find the TRIM deflection angle and aerodynamic coefficients. If a fin deflection of zero is not one of the ten deflections that results from this process, the trim axial force and drag coefficients will be in error for deflections in between the positive and negative deflections with the lowest magnitude. For example, consider the default values of DELMIN=-25 and DELMAX=+20. The code will interpolate between results generated with the following deflections:

$\delta = -25, -20, -15, -10, -5, 0, +5, +10, +15, +20$

For a symmetric vehicle at zero angle of attack, the control deflection for trim will be  $\delta=0$ , and a good result will be obtained this value is included in the interpolation. Now, assume DELMIN=-45 and DELMAX=+45. The code will generate the following deflections for interpolation:

$\delta = -45, -35, -25, -15, -5, +5, +15, +25, +35, +45$

Since  $\delta=0$  is not in the preceding list, the code will interpolate between the  $\delta=-5$  and  $\delta=+5$  cases for the  $\delta=0$  result. This will result in an incorrect trim drag since the drag at  $\delta=-5$  and  $\delta=+5$  is equal at  $\alpha=0$  and higher than the actual trim drag at  $\delta=0$ .

### ***NAMelist TRIM***

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
SET	-	Fin set to be used for trim	-	1.
PANL1	-	.TRUE. if panel to be used	-	.FALSE.
PANL2	-	.TRUE. if panel to be used	-	.FALSE.
PANL3	-	.TRUE. if panel to be used	-	.FALSE.
PANL4	-	.TRUE. if panel to be used	-	.FALSE.
PANL5	-	.TRUE. if panel to be used	-	.FALSE.
PANL6	-	.TRUE. if panel to be used	-	.FALSE.
PANL7	-	.TRUE. if panel to be used	-	.FALSE.
PANL8	-	.TRUE. if panel to be used	-	.FALSE.
DELMIN *	-	Minimum negative deflection	deg	-25.
DELMAX *	-	Maximum positive deflection	deg	+20.
<b>NOTE:</b> * If only one of DELMIN or DELMAX is specified, the other will default to 0.				

Table 20. NAMELIST TRIM Variables

### 3.1.9 Namelist INLET - Axisymmetric and 2-Dimensional Inlet Geometry

This namelist is used to model the inlet and diverter geometry. Axisymmetric, two-dimensional side mounted, and two-dimensional top mounted inlets can be described. The inlets may be covered or uncovered and oriented in any position about the missile body. Inlet normal force, pitching moment, side force, yawing moment, and axial force are calculated. The methods are valid for subsonic, transonic, and supersonic speeds. **Table 21** shows the INLET namelist inputs, and **Figures 13, 14, 15, and 16** show the inlet/diverter geometry for each type of inlet. The inlets may have a boundary layer diverter, be conformal (diverter height HDIV=0), or be semi-submerged (diverter height HDIV<0). The methods used for the inlets are the same regardless of whether the inlet has a diverter or is semi-submerged, but they are **not** applicable to chin inlets. The variable HDIV is used to determine whether a diverter exists. **Figure 16.** shows examples of two-dimensional and axisymmetric inlets that are conformal or semi-submerged.

- Inlet roll orientation uses the same convention as the fin panel roll orientation.
- Inlet height and width or inlet diameter is input at five axial locations described in **Figures 13, 14, and 15**:
  - 1) leading edge or tip
  - 2) cowl lip leading edge
  - 3) midbody start
  - 4) boattail start
  - 5) boattail end
- If the inlet is covered (COVER=.TRUE.), no flow is allowed into the inlet. The inlet is plugged between stations 1 and 2, flush with the inlet face.

Inlet additive drag or spillage drag can be calculated for external compression inlets operating at off-design conditions ( $M_\infty < M_{\text{design}}$ ) for Mach numbers greater than 1. Whenever flow spillage occurs, the mass flow ratio is less than one, and additive forces are generated on the deflected streamtube captured by the inlet. If the inlet operates on-design, the ramp shock lies on the inlet face and on the cowl lip. In these cases, the maximum mass flow ratio is one (zero spillage) and the minimum additive forces are zero.

- If the inlet is covered (COVER=.TRUE.), the additive drag calculations will be skipped.
- If ADD=.FALSE., or is not input the additive drag calculations will be skipped.
- Mass flow ratio (MFR) must be specified for each freestream Mach number or velocity given in namelist FLTCON. For Mach numbers less than 1, dummy values must be input for MFR. The user must be careful to match these inputs with the proper freestream conditions.
- The additive drag is calculated at zero angle of attack and assumed to remain constant for all angles of attack.



### **NAMelist INLET**

VARIABLE NAME	ARRAY SIZE	DEFINITION	UNITS	DEFAULT
NIN	-	Number of inlets (maximum 20)	-	-
INTYPE	-	Type of inlet and numerical equivalent: 2DSIDE <or 1.> (side mounted 2-D) AXI <or 2.> (axisymmetric) 2DTOP <or 3.> (top mounted 2-D)	-	-
XINLT	-	Longitudinal distance from nose tip to inlet leading edge	L	-
XDIV	-	Longitudinal distance from inlet leading edge to diverter leading edge	L	-
HDIV	-	Diverter height at leading edge. HDIV=0 defines conformal inlet. HDIV<0 defines semi-submerged inlet	L	-
LDIV	-	Length of diverter	L	-
PHI	20	Inlet roll orientations measured clockwise from top vertical center looking forward	deg	-
X *	5	Inlet longitudinal positions relative to inlet leading edge	L	-
H *	5	Inlet heights at the longitudinal positions. Not required if INTYPE=AXI	L	-
W *	5	Inlet widths at the longitudinal positions if INTYPE=2DTOP or 2DSIDE. Inlet diameters if INTYPE=AXI	L	-
COVER	-	Flag for covered inlet: .TRUE. (inlet covered) .FALSE. (inlet open)	-	.FALSE.
RAMP	-	External compression inlet ramp angle	deg	-
ADD	-	Flag for inlet additive drag: .TRUE. (calculate additive drag) .FALSE. (do not calculate)	-	.FALSE.
MFR	20	Mass flow ratio for each Mach number in namelist \$FLTCON. 0.0 <MFR <1.0. Only required if ADD=.TRUE.	-	-
<p>Note:* Specify X, H and W at five inlet locations as shown in <b>Figure 13 – 15.</b> (1) leading edge, (2) cowl lip, (3) midbody start (4) boattail start, and (5) boattail end. The inlet must be boattailed, meaning <math>H(5)*W(5) &lt; H(4)*W(4)</math> for 2D inlets, or <math>W(5) &lt; W(4)</math> for axisymmetric inlets.</p>				

Table 21. NAMelist INLET Variables

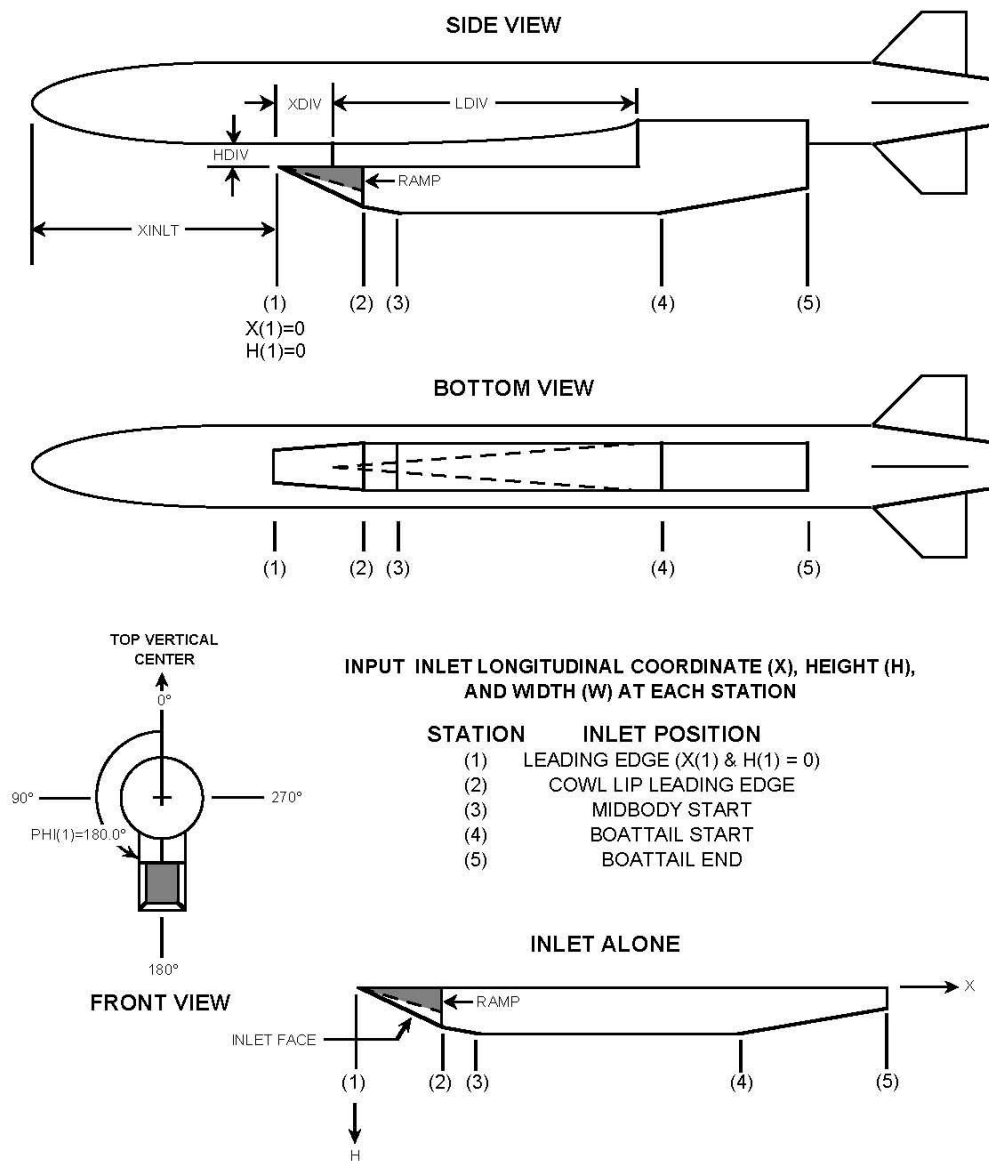


Figure 13. Top-Mounted 2-D Inlet/Diverter Geometry

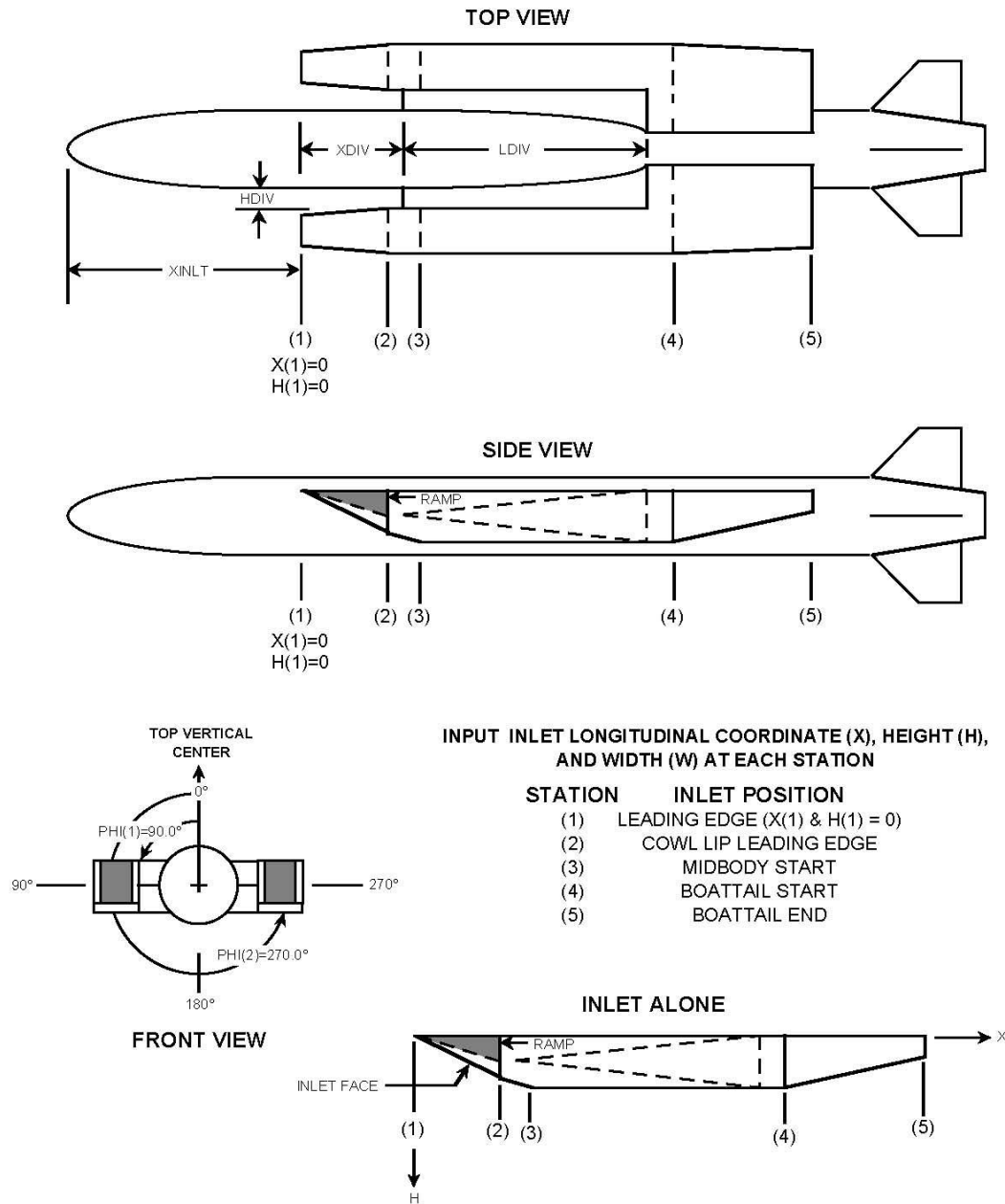
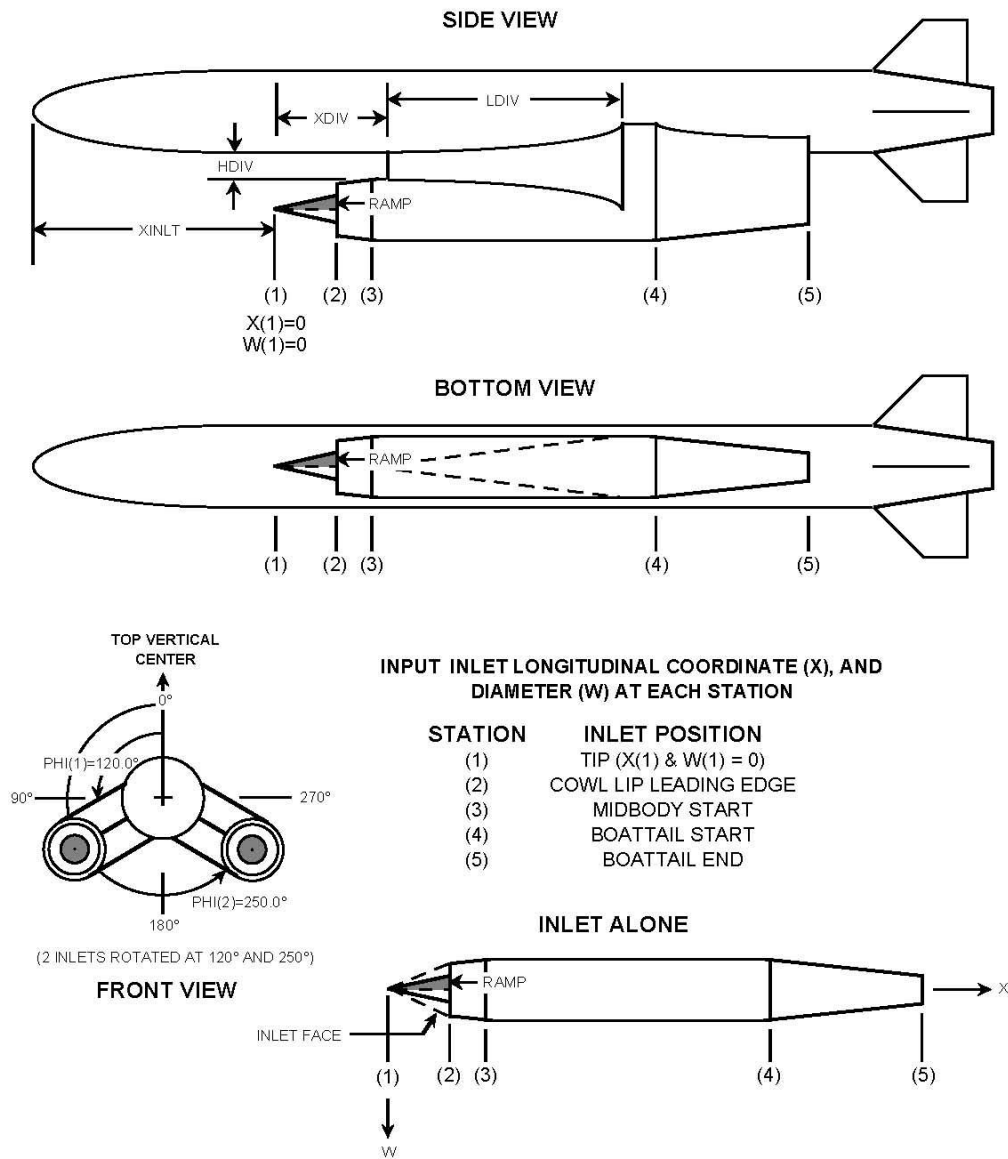


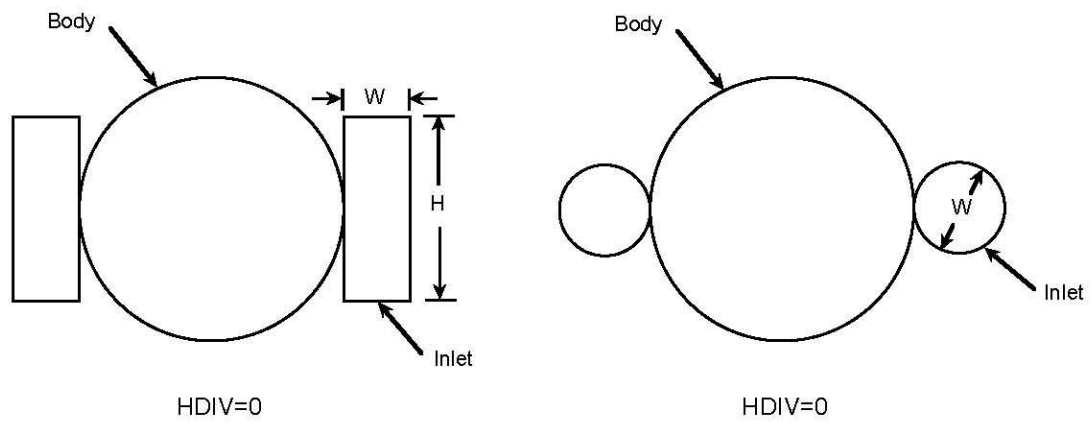
Figure 14. Side-Mounted 2-D Inlet/Diverter Geometry



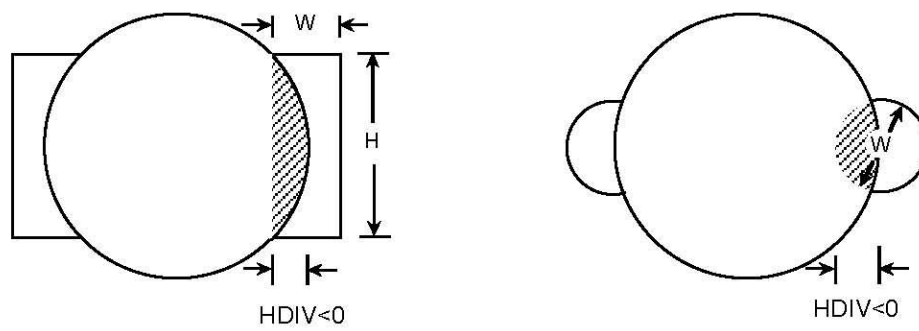
**NOTES:**

- INLET ROLL ORIENTATION IS SAME CONVENTION AS FIN ROLL ORIENTATION.
- RAMP IS THE EXTERNAL COMPRESSION CONE HALF-ANGLE (SHOWN SHADED IN THE SIDE VIEW)
- HEIGHT OF THE DIVERTER IS SPECIFIED AT THE DIVERTER LEADING EDGE
- THE DIVERTER WIDTH IS EQUAL TO THE INLET DIAMETER AT LDIV
- IF INLET IS COVERED (COVER=.TRUE.) A PLUG IS PLACED BETWEEN STATIONS 1 AND 2 FLUSH WITH THE INLET FACE

Figure 15. Axisymmetric Inlet/Diverter Geometry



### Conformal Inlets



### Semi-Submerged Inlets

Figure 16. Geometry Definition for Conformal and Semi-Submerged Inlets

### 3.1.10 Namelist EXPR - Experimental Data Substitution

This namelist is used to substitute experimental data for the theoretical data generated by the program. The variables to be input are shown in **Table 22**. Use of namelist EXPR does not stop the program from calculating theoretical data, but rather the experimental data is used in configuration synthesis, and it is the experimental data that is used for the component aerodynamics for which it is input.

Experimental data may be substituted for any configuration component or partial configuration. Experimental data is input at a specific Mach number. When using namelist EXPR, the case must be run at the Mach number for which you are substituting experimental data. The experimental data being input may have different reference quantities and a different center of gravity location than the case being run.

Experimental data input for a fin alone is input as panel data, not as total fin set data. The user should note that experimental data for fin alone  $C_{M\alpha}$  is not used in the configuration synthesis process. Instead fin alone  $C_{N\alpha}$  (the experimental value if input) is used to determine the fin contribution to  $C_{M\alpha}$  during configuration synthesis. If body alone experimental data and body-fin experimental data are input for the same case the body data is ignored in configuration synthesis. If experimental  $C_{M\alpha}$  data is input for a body + 1 fin set for a multi-fin set configuration, the calculated contributions to  $C_{M\alpha}$  of the other fin sets are added to the experimental data.

Since the experimental namelist forms the basis for configuration incrementing, the lateral directional coefficients are included to allow for sideslip cases. These coefficients are input the same as the longitudinal coefficients. However, if the lateral directional coefficients are input, the lateral directional beta derivatives will not be computed our output.

The following rules apply to the use of namelist EXPR.

- It is assumed that the coefficients in EXPR are for the same sideslip and/or aerodynamic roll as the case being run.
- Separate namelist EXPR must be specified for each Mach number.
- Separate namelist EXPR must be specified for each partial configuration for which experimental data is to be input, (i.e., body, body + 1 fin set, etc)
- Separate namelist EXPR must be specified for each reference quantity change.

Example:

The user has experimental data available for a body + 2 fin set configurations and is interested in the effects of adding a booster containing a third fin set. he would then use namelist EXPR to input the experimental data. When the configuration is synthesized, it would use the experimental data for body + 2 fin sets and theoretical data for fin set three.

### ***NAMelist EXPR***

<b>VARIABLE NAME</b>	<b>ARRAY SIZE</b>	<b>DEFINITION</b>	<b>UNITS</b>	<b>DEFAULT</b>
MACH	-	Mach number	-	-
NALPHA	-	Number of angles of attack	-	-
ALPHA	100	Angle of attack for data	deg	-
SREF	-	Reference area for data	L <sup>2</sup>	*
LREF	-	Longitudinal reference length for data	L	**
LATREF	-	Lateral reference length for data	L	LREF
XCG	-	Longitudinal C.G. for data	L	0.
ZCG	-	Vertical C.G. for data	L	0.
CONF	-	Configuration for which data is to be supplied and numerical equivalent: BODY <or 1.> (body) F1 <or 2.> (fin set 1) F2 <or 3.> (fin set 2) F3 <or 4.> (fin set 3) F4 <or 5.> (fin set 4) BF1 <or 6.> (body +1 fin set) BF12 <or 7.> (body +2 fin sets) BF123 <or 8.> (body +3 fin sets) BF1234 <or 9.> (body +4 fin sets)	-	-
CN	100	C <sub>N</sub> data vs alpha	-	-
CM	100	C <sub>M</sub> data vs alpha	-	-
CA	100	C <sub>A</sub> data vs alpha	-	-
CY	100	C <sub>Y</sub> data vs alpha	-	-
CSN	100	C <sub>n</sub> data vs alpha	-	-
CSL	100	C <sub>l</sub> data vs alpha	-	-
<b>Notes:</b> * Default is maximum body cross-sectional area. If no body is input, default is maximum fin panel area. ** Default is maximum body diameter. If no body is input, default is fin panel mean geometric chord.				

Table 22. NAMelist EXPR Variables

## 3.2 CONTROL CARD INPUTS

Control cards are one line commands which select program options. Although they are not required inputs, they permit user control over program execution and the types of output desired. Control cards enable the following:

- Outputting intermediate calculations (PART, BUILD, PRESSURES, PRINT AERO, PRINT GEOM, PLOT, NAMELIST)
- Selecting the system of units to be used (DIM, DERIV)
- Defining multiple cases, permitting the reuse of previously input namelist data or deleting namelists of a prior case (SAVE, DELETE, NEXT CASE)
- Adding case titles or comments to the input file and output pages (\*, CASEID)
- Limits the calculations to longitudinal aerodynamics (NO LAT)

### 3.2.1 Control Card - General Remarks

There is no limit to the number of control cards that can be present in a case. If two or more control cards contradict each other, the last control card input will take precedence. All control cards must be input as shown, including any blanks. Control cards can start in any column but they cannot be continued to a second card. Misspelled cards are ignored. Control cards can be located anywhere within a case.

Once input, the following control cards remain in effect for all subsequent cases, but may be subsequently changed in later cases:

DIM FT	DIM IN	DIM CM	DIM M
DERIV DEG	DERIV RAD	HYPER	NOGO
NO LAT	PLOT	SOSE	

The following control cards are effective only for the case in which they appear:

BUILD	CASEID	DAMP	DELETE
NAMELIST	PART	PRESSURES	PRINT AERO
PRINT GEOM	SAVE	SPIN	TRIM

The following control card is the **ONLY** control card that may be saved:

NACA



### 3.2.2 Control Card Definitions

Available control cards are summarized as follows:

#### BUILD

This control card instructs the program to print the results of a configuration build-up. All configurations which can be built from the components defined will be synthesized and output, including isolated data (e.g., body alone, fin alone, etc.). Component build-up data is not provided if the TRIM option is selected. This control card is effective only for the case in which it appears.

#### CASEID

A user supplied title to be printed on each output page is specified. Up to 72 characters can be specified (card columns 8 to 80). This control card is effective only for the case in which it appears.

#### DAMP

When DAMP control card is input dynamic derivatives are computed and the results output for the configuration. The longitudinal (pitch rate) derivatives are non-dimensionalized by the quantity  $(q \cdot L_{REF} / 2 \cdot V_{INF})$ . The lateral-directional (roll rate, yaw rate) derivatives are non-dimensionalized by the quantities  $(p \cdot L_{ATREF} / 2 \cdot V_{INF})$  and  $(r \cdot L_{ATREF} / 2 \cdot V_{INF})$  respectively. Dynamic derivatives for configuration components or partial configurations may be output using the PART or BUILD control cards respectively. This control card is effective only for the case in which it appears. Damping derivatives are NOT computed if PHI or BETA are non-zero.

#### DELETE name1,name2

This control card instructs the program to ignore a previous case namelist input that was retained using the SAVE control card. All previously saved namelists with the names specified will be purged from the input file. Any new inputs of the same namelist will be retained. At least one name (name1) must be specified. The DELETE control cards are effective only for the case in which they appear.

#### DIM IN, DIM FT, DIM CM, or DIM M

This control card sets the system of units for the user inputs and program outputs. The four options are inches (DIM IN), feet (DIM FT), centimeters (DIM CM), and meters (DIM M). The default system of units is feet. Once the system of units has been set, it remains set for all subsequent cases of the "run". All units for geometric length are associated with DIM while others listed below in **Table 23** are associated with whether DIM is in the English or Metric system.

<b>Namelist</b>	<b>VAR</b>	<b>DIM IN</b>	<b>DIM FT</b>	<b>DIM CM</b>	<b>DIM M</b>
FLTCON	REN	1/feet	1/feet	1/meter	1/meter
FLTCON	ALT	feet	feet	meter	meter
FLTCON	VINF	feet/sec	feet/sec	meter/sec	meter/sec
FLTCON	PINF	pounds/feet <sup>2</sup>	pounds/feet <sup>2</sup>	Newtons/meter <sup>2</sup>	Newtons/meter <sup>2</sup>
FLTCON	TINF	Rankine	Rankine	Kelvin	Kelvin
REFQ	ROUGH	inches	inches	centimeters	centimeters

Table 23. Units Required for Flight Condition Variables and Surface Roughness

### DERIV DEG or DERIV RAD

All output derivatives are set to either degree (DERIV DEG) or radian (DERIV RAD) measure. The default setting is degree. The derivative units can be changed between cases by inputting additional DERIV cards.

### HYPER

This control card causes the program to select the Newtonian flow method for bodies at any Mach number above 1.2. HYPER should normally be selected at Mach numbers greater than 6. HYPER causes carryover effects to be ignored for all supersonic Mach numbers.

### NACA

This card defines the NACA airfoil section designation (or supersonic airfoil definition). Note that if airfoil coordinates and the NACA card are specified for the same aerodynamic surface, the airfoil coordinate specification will be used. Therefore, if coordinates have been specified in a previous case and the SAVE option is in effect, the saved namelist must be deleted or the namelist variable SECTYP must be changed for the NACA card to be recognized for that aerodynamic surface. The airfoil designated with this card will be used for all segments and panels of the fin set.

The form of this control card and the required parameters are as follows:

<u>Card Column(s)</u>	<u>Input(s)</u>	<u>Purpose</u>
1 thru 4	NACA	The unique letters NACA designate that an airfoil is to be defined
5	Any delimiter	
6	1,2,3, or 4	Fin set number for which the airfoil designation applies
7	Any delimiter	
8	1,4,5,6,S	Type of NACA airfoil section; 1-series (1), 4-digit (4), 5-digit (5), 6-series (6), or supersonic (S)
9	Any delimiter	
10 thru 80	Designation	Input designation (see Table 6); columns are free-field (blanks are ignored)

Only fifteen (15) characters are accepted in the airfoil designation. The vocabulary consists of the following characters:

0    1    2    3    4    5    6    7    8    9    A    ,    =    .    -

Any characters input that are not in the vocabulary list will be interpreted as the number zero (0). **Table 18** details the restrictions on the NACA designation.

### NAMELIST

This control card instructs the program to print all namelist data. This is useful when multiple inputs of the same variable or namelist are used. This control card is effective only for the case in which it appears.

### NEXT CASE

This card indicates termination of the case input data and instructs the program to begin case execution. It is required for multiple case "runs". This card must be the last card input for the case.

### NOGO

This control card permits the program to cycle through all of the input cases without computing configuration aerodynamics. It can be present anywhere in the input stream and only needs to appear once. This option is useful for performing error checking to insure all cases have been correctly set up.

### NO LAT

This control card inhibits the calculation of the lateral-directional derivatives due to sideslip angle, and the roll rate and yaw rate derivatives if the control card DAMP is selected. Large savings in computation time can be realized by using this option. This option is automatically selected when using TRIM.

### PART

This control card permits printing of partial aerodynamic output, such as a summary of the normal force and axial force contributors. Partial output of the configuration synthesis methods is only provided if the TRIM option is not selected. Use of this card is equivalent to inputting all PRINT AERO and PRINT GEOM control cards. This control card is effective only for the case in which it appears.

### PLOT

A data file for use with a post-processing plotting program is provided when this control card is used. A formatted file is written to tape unit 3 ("for003.dat").

## PRESSURES

This control card instructs the program to print the body and fin alone pressure coefficient distributions at supersonic speeds. Only pressure data to 15 degrees angle of attack for bodies and at zero angle of attack for fins are printed. The body pressure output at positive angle of attack is written to tape unit 10 ("for010.dat"). The fin pressure output is written to tape unit 11 ("for011.dat"). The body pressure output and local Mach number at zero angle of attack is written to tape unit 12 ("for012.dat"). This control card is effective only for the case in which it appears.

## PRINT AERO name

This control card instructs the program to print the incremental aerodynamics for "name", which can be one of the following:

BODY	for body aerodynamics
FIN	for all fin set aerodynamics
SYNTHS	for configuration synthesis aerodynamics
TRIM	for trim/untrimmed aerodynamics
BEND	for panel bending moments
HINGE	for panel hinge moments
INLET	for inlet aerodynamics

All options are automatically selected when the control card PART is used. Details of the output obtained with these options are presented in **Section 4.2**. The PRINT AERO control cards are effective only for the case in which they appear.

## PRINT GEOM name

This control card instructs the program to print the geometric characteristics of the configuration component "name", which can be one of the following:

BODY	for body geometry
FIN	for all fin set geometries
INLET	for inlet geometry

If PRINT GEOM BODY is selected and the Mach number is greater than 1.2, the body contour coordinates (X,R) used by the program are written to tape unit 9 ("for009.dat"). This contour will contain many additional points in between the user specified input coordinates, and is useful for verifying that the DISCON values have been properly entered.

All options are automatically selected when the control card PART is used. The PRINT control cards are effective only for the case in which they appear.

## SAVE

The SAVE card saves namelist inputs from one case to the following case but not for the entire run (only effective for the case in which it appears). This permits the user to build-up or change a complex configuration, case-to-case, by adding new namelist cards without having to re-input namelist cards of the previous case. To reinitialize a namelist that has been saved, the namelist must first be deleted using the DELETE control card. If the SAVE card is used, the

total number of namelists read, including repeat occurrences of the same namelist name from all previous SAVES, must not exceed 300. A warning message is output when this limit would be exceeded. For more information see the comments on SAVE listed under “typical set-up” below.

## SOSE

The presence of this control card selects the Second-Order Shock Expansion method for bodies at supersonic speeds. The code will attempt to use the Van Dyke Hybrid theory first for any Mach number greater than 1.2. If the body has a blunted nose or the Van Dyke Hybrid method fails, the program will use the Second-Order Shock Expansion method. Use of the control card SOSE forces the program to use this method. It is generally recommended if the Mach number is higher than 2.0.

## SPIN

When the SPIN control card is input, spin and magnus derivatives are computed for body alone. If the configuration being run is a body + fin sets, the spin derivatives are still computed for body alone. As the SPIN card affects only damping derivatives, the DAMP card must also be specified to see any results in the output file. A PART or BUILD card must be input for body alone derivatives to be printed out. This control card is effective only for the case in which it appears.

The following data will be computed and output:

CLLP	Rolling moment due to roll rate (roll damping)
CYP	Side force due to roll rate
CNP	Is a polynomial fit of CNP1, CNP3, and CNP5 given by: $\text{CNP} = \text{CNP1} * \sin(\text{ALPHA}) + \text{CNP3} * (\sin(\text{ALPHA}))^3 + \text{CNP5} * (\sin(\text{ALPHA}))^5$
CNP1	Zero yaw Magnus coefficient derivative per sin alpha
CNP3	Cubic Magnus moment coefficient per (sin alpha)**3
CNP5	Quintic Magnus moment coefficient per (sin alpha)**5
CNPY5	5 degree secant slope of Magnus moment coefficient derivative (at five degree yaw) per sin alpha

Table 24. Magnus derivatives calculated with SPIN Control Card

## TRIM

This control card causes the program to perform a trim calculation. Component buildup data cannot be dumped if TRIM is selected. The use of this control card is the same as if the namelist TRIM was included except that the defaults for namelist TRIM are used. This control card is effective only for the case in which it appears.

## COMMENT CARDS

Any card with an asterisk (\*) in Column 1 will be interpreted as a comment card. This permits detailed documentation of case inputs.

### 3.3 TYPICAL CASE SET-UP

**Figure 17** schematically shows how Missile Datcom inputs are structured. This example illustrates a multiple case job in which case 2 uses part of the case 1 inputs. This is accomplished through use of the SAVE control card. Case 1 is a body-wing-tail configuration; partial output, component buildup data, and a plot file are created. Case 2 uses the body and tail data of case 1 (the wing is deleted using DELETE), specifies panel deflection angles and sets the data required to trim.

There is no limit to the number of cases that can be "stacked" in a single run, provided that no more than 300 namelist inputs are "saved" between cases. If a SAVE control card is not present in a case, all previous case inputs are deleted.

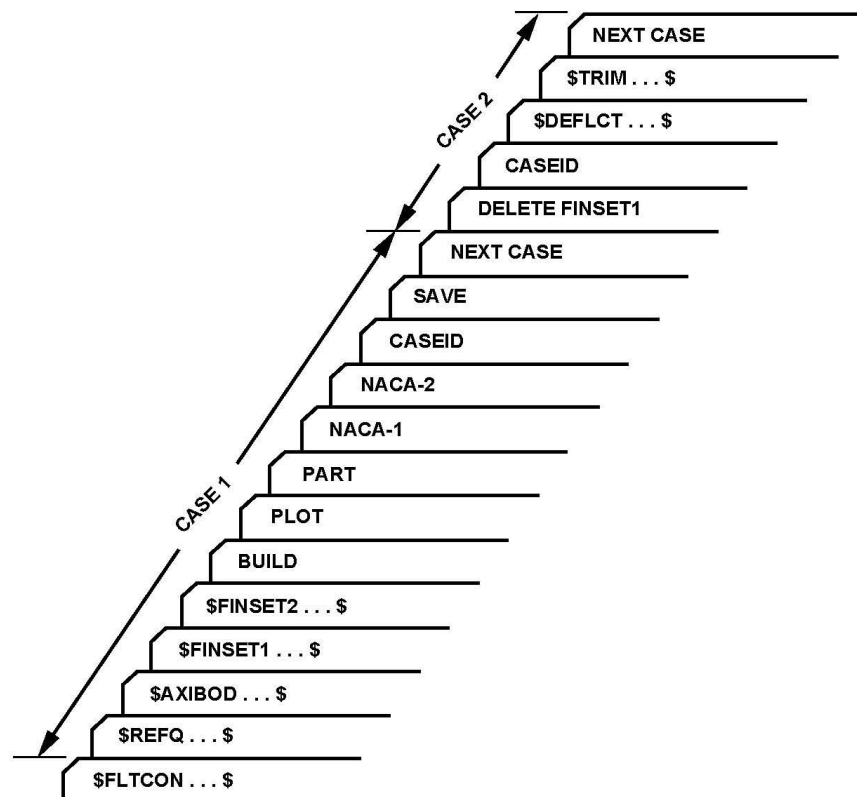


Figure 17. Typical "Stacked" Case Set-up.

## 4.0 OUTPUT DESCRIPTION

This section describes the types of output available from the code. In many cases the available output is user selectable, that is, it is not normally provided and must be specifically requested using a specialized control card. This feature permits the user to tailor the code output to fit his particular application without extensive reprogramming. This allows him to find the output that he is interested in without having to wade through output that does not interest him.

The following three types of output are available from the code:

- Nominal output - This output is always provided by the code and consists of output from the input error checking module (CONERR), a listing of the inputs for each case, and the final aerodynamic results for the configuration.
- Partial output - This output details the configuration geometry and the intermediate aerodynamic calculations. Special control cards are available so that the user can select the quantity and types of output desired.
- External data files - This output permits the user to create external data files which can be used in post-processing programs, such as plotting or trajectory programs. Both fixed and user defined format data files can be created with the addition of simple control cards.

The remainder of the section describes each of these output data. Examples of each output page are also included and were created from the example problems, described in Appendix A, which can be used as a model for setting up another, similar configuration or be used as a means to check the proper operation of the code. A complete listing of the output files is given in **Table 1**.

### 4.1 NOMINAL OUTPUT

Without the use of any program options the code will provide three types of output. First, an analysis by the input error checking routine is provided. It lists all input cards provided by the user and identifies any input errors detected. Second, a listing of all input cards, grouped by case, are provided; included in this output is an error analysis from the major input error routine MAJERR. Finally, the total configuration aerodynamics are provided in summary form; one page of aerodynamic output is supplied for each Mach number specified. The MAJERR results and the total configuration aerodynamics results are listed in succession for each case.

#### 4.1.1 Input Error Checking

The purpose of the input error checking module is to provide single pass error checking of all inputs. If an error is detected, it is identified and an appropriate error message provided. The error messages are designed to be self-explanatory. In some cases, errors are automatically corrected by the routine, although the routine was not designed to be a comprehensive error correction utility.

The following errors are automatically corrected by the code:

- No terminating comma on a namelist input card
- No terminating "\$" or "\$END" on a namelist input ("&" on IBM systems)
- No terminating NEXT CASE for the case inputs for single case or last case inputs.

Errors detected by the error checking routine are considered either "FATAL" or "NON-FATAL". A "FATAL" error is one which will cause the code to terminate execution abnormally; examples of "FATAL" errors include incorrect spelling of any namelist name, incorrect spelling of any variable name, and any drastic input error in a namelist input, such as leaving out an equals sign in a constant definition. All "FATAL" errors are clearly identified on the output. A "NON-FATAL" error is one which will not cause the program to terminate execution. An example of a "NON-FATAL" error is leaving off the decimal point on numeric constants. Missile Datcom inputs are either REAL or LOGICAL regardless of the variable name assigned.

An example output from CONERR is shown in **Figure 18**. This figure illustrates the array of input errors checked by CONERR. Several additional features of the output are as follows:

- All user defined input cards are assigned a sequential "line number". This serves to identify user inputs from the code generated inputs (all code-created input cards are not identified with a "line number"). This scheme also permits the user to quickly identify those input cards in error so that efficient correction of input errors can be performed.
- All input cards are listed as input by the user. To the right of each input card is a listing of any errors encountered in processing that card. If no such error message appears then the input was interpreted as being correct.
- In many cases alphanumeric constants are available (see **Table 2**). Hence the user does not need to memorize a numeric scheme of "flags". Since some computers do not recognize alphanumeric constants as namelist constants, they are automatically converted by the code to their numeric equivalent. A message is printed to identify the substitutions performed.

In order to permit column independent inputs the code will automatically adjust some of the input cards to begin in columns 1 or 2. All control cards will be automatically shifted to start in column 1; all namelists which begin in column 1 will be shifted to column 2. If any input card cannot be shifted to conform to this scheme, an error message will be produced. As a general rule, column 80 of namelist inputs should be left blank so that the code can shift the card image, if necessary.

#### 4.1.2 Listing of Case Input Data

**Figure 19** shows the first page of outputs for a case without CONERR detected errors. Then **Figure 20** shows the next page of output which lists all input cards for the case (down to the NEXT CASE control card). If the input for a case is from a previous case (through use of the SAVE control card) only the new case inputs are listed. All saved inputs are not repeated in subsequent case input summaries.



After the case data have been read, the data set-up for the case is analyzed by the case major error checking module (MAJERR). The purpose of this second error checking is to insure that the data input, although syntax error free, properly defines a case to be run. Examples of errors detected in MAJERR include valid flight condition inputs, valid reference condition inputs, and that geometry has been defined. In most cases errors detected by MAJERR are corrected with assumed defaults. If any MAJERR error message is produced, the user should verify the "fix-up" taken by the code. In some cases a "fix-up" is not possible; an appropriate error message and a suggestion for correcting the error is provided. If a "fix-up" is not possible the case will not run.

#### 4.1.3 Case Total Configuration Aerodynamic Output Summary

As shown in **Figure 21**, the total configuration aerodynamics are provided in compact form for easy review. The aerodynamics are summarized as a function of angle of attack (ALPHA) in the user specified system of units, and are given in the body axis system. The nomenclature is as follows:

CN	- Normal force coefficient
CM	- Pitching moment coefficient
CA	- Axial force coefficient
CY	- Side force coefficient
CLN	- Yawing moment coefficient
CLL	- Rolling moment coefficient
CNA	- Normal force coefficient derivative with ALPHA
CMA	- Pitching moment coefficient derivative with ALPHA
CYB	- Side force coefficient derivative with BETA
CLNB	- Yawing moment coefficient derivative with BETA
CLLB	- Rolling moment coefficient derivative with BETA
CL	- Lift coefficient
CD	- Drag coefficient
CL/CD	- Lift to drag ratio
XCP	- Center of pressure position from the moment reference center divided by reference length, positive values for c.p. forward of the moment reference point.

All coefficients are based upon the reference areas and lengths specified at the top of the output page. The derivatives CNA and CMA are computed by numeric differentiation of the CN and CM curves, respectively; precise derivatives are only obtained when the angle of attack range specified is narrow. The derivatives CYB, CLNB and CLLB are determined by perturbing the sideslip angle by one degree, recalculating the configuration forces and moments, and then differencing with the user specified orientation. Hence, the longitudinal and lateral derivatives may not be numerically identical for those conditions which should produce identical results if they were both calculated by the same method.

A significant decrease in computational time is realized when the calculation of lateral-directional derivatives are suppressed using the control card NO LAT. For these cases, the CYB, CLNB, and CLLB data fields are filled with blanks.

When selecting TRIM, the output is provided in a form similar to **Figure 22**. When running a trim case the derivatives due to ALPHA and BETA are not available. The panels

which were deflected to trim the configuration are indicated by the "VARIED" citation next to them.

The format for the values of the numbers in the printed output has been assumed based on typical magnitudes for missile aerodynamic coefficients. In some cases, a user specified reference area and/or length will cause the results to underflow or overflow the format selected. For these cases the user should adjust his reference quantities by powers of ten to get the data to fit the format specified.

## 4.2 PARTIAL OUTPUT

Partial output consists of geometry calculation details, intermediate aerodynamic results, or auxiliary data, such as pressure distributions. Each of these output types are printed through the addition of control cards input for each case. In all cases, partial output requested for one case is not automatically selected for subsequent cases, and the control cards must be re-input. This permits the user to be selective on the amount and types of output desired.

A special control card PART permits the user to request all geometric and aerodynamic partial output. Due to the amount of output produced, this option should be used sparingly or when details of the calculations are desired.

The following paragraphs describe the output received when partial output is requested.

### 4.2.1 Geometric Partial Output

Details of the geometry are provided when the PART or PRINT GEOM control cards are included in the case inputs. **Figure 23** shows the output created when the PRINT GEOM BODY control card is used. Detailed are the results of the geometric calculations for the body. Included are such items as planform area, surface (wetted) area, and the mold line contour.

If fins are present on the configuration, two types of fin geometry data are produced when PRINT GEOM FIN1 or PART is requested. As shown in **Figure 24**, the description of the panel airfoil section is provided. Following that, shown in **Figure 25**, is a summary of the major geometric characteristics of such planform; note that fin planform geometry data is given for one panel of each fin set, since it is assumed that each fin of a fin set is identical. If a panel is made up of multiple segments, the geometric data is provided by panel segment (each segment is assigned a number starting at the root). Total panel set of characteristics is also provided. This total panel data represents an equivalent straight-tapered panel, which is used for most of the aerodynamic calculations. The thickness-to-chord ratio shown for each segment is that value at the segment root; for the total panel, it is an "effective" value.

If an airbreathing inlet is specified the output is similar to that in **Figure 27**. This output reflects the user input definition for the inlet design specified. It is provided if the PRINT GEOM INLET or PART control cards are included in the input case.

### 4.2.2 Aerodynamic Partial Output

The output on the configuration aerodynamics is most extensive when PRINT AERO or PART is specified. Output is created for the body and each fin set on the configuration. In addition, for any subsonic/transonic Mach number (less than 1.4) an analysis by the Airfoil

Section Module is made, which involves a potential flow analysis of the airfoil section using conformal mapping. If a configuration has inlets additional partial output is included to summarize the inlet external aerodynamics.

If base-jet plume interaction calculations are specified (BASE=.TRUE. in namelist AXIBOD), then there will be one or two separate pages of output. **Figure 28** shows an example of the first page of output. This page will always be printed if BASE=.TRUE. The base pressure coefficient, axial force coefficient, and freestream pressure and temperature ratios are shown versus angle of attack. Also, the incremental forces and moments due to separation are shown versus angle of attack. If extrapolation of the base pressures and separation conditions database occurs, a warning message is printed explaining what input variable required extrapolation. A second page of output containing the boattail separation parameters will be printed if there are any fins on the missile boattail. The separation location aft of the nose and the Mach cone angle are shown versus angle of attack for each panel on the fin set. This output is provided if the PRINT AERO BODY or PART control card is input.

The protuberance partial output is printed if PRINT AERO BODY or PART is used. This output will only be shown if the namelist PROTUB is present in the input file. **Figure 29** is an example of the protuberance output. Protuberance type, location, number, and axial force coefficient are listed for each protuberance set. The total axial force coefficient or zero lift drag coefficient is printed at the bottom of the page.

As shown in **Figure 30**, the body alone partial aerodynamic output for normal force lists the axial force contributors, potential normal force (CN-POTENTIAL), viscous normal forces (CN-VISCOUS), potential pitching moment (CM-POTENTIAL), viscous pitching moment (CM-VISCOUS), and the crossflow drag coefficient (CDC). The cross-flow drag proportionality factor at subsonic and transonic speeds is also given for reference. These data are similar to that obtained for elliptical bodies.

**Figure 31** details the fin normal force calculations by fin set. Each panel's contribution to the configuration normal force is described. The column titled CN-POTENTIAL is the potential contribution and the column titled CN-VISCOUS is the viscous contribution. Their sum is given in the column titled CN-TOTAL. CNAA is the nonlinear variation of normal force due to angle of attack and ALPHA EQUIV is the panel angle of attack due to its roll position on the body. **Figure 32** illustrates the fin axial force contributors and **Figure 31** presents an example of the fin pitching moment contributors.

The analysis by the Airfoil Section Module is provided in a format similar to **Figure 33**. If any Mach number specified produces supersonic flow on the airfoil surface, the message "CREST CRITICAL MACH NUMBER EXCEEDED" will be printed; approximation of the airfoil section data is then assumed. These fin aerodynamic increments are repeated for each fin set on the configuration. Note that the Airfoil Section Module assumes that the panels have sharp trailing edges. Any panel input with a non-sharp trailing edge will have its aerodynamic characteristics set as though the airfoil was "ideal". This assumption is approximate for preliminary design.

**Figure 34** shows the aerodynamic output available when inlets are specified on the configuration. It is provided when PRINT AERO INLET or PART is specified in the case inputs. The aerodynamics summarized for inlets can include additive drag results if the user input the additive drag calculation flag. The maximum mass flow ratio is printed at the bottom of the page

if the additive drag is calculated. If additive drag cannot be calculated, a warning message is printed.

After the aerodynamic details for each component of the configuration are output, the aerodynamic calculations for the synthesis of the complete configuration follows. For the example case, fin set 1 results would be followed by fin set 2 results for each of the following outputs:

- "FIN SET IN PRESENCE OF THE BODY" - This summarizes the aerodynamic incrementals of the most forward set of fins with the influence of the body. **Figure 35 and 36** presents the example of this output. The left-most six columns include the effect of body-on-fin component interference. The right-most columns represent the contribution to each panel to configuration aerodynamics, and include the effect of body-on-fin interference, these values are, in effect, individual panel loads. The panel characteristic values included are "AEQn" (the panel equivalent (local) angle of attack) and "CNn" (the panel normal force coefficient). The sign convention is as follows: a positive panel normal force, hence, equivalent angle of attack, produces a negative roll moment. Therefore, panels on the right side of the configuration will produce loads and angles of attack opposite in sign to those on the left side of the configuration even though they produce the same physical force loading.
- " CARRYOVER INTERFERENCE FACTORS " - This page of partial output summarizes the carryover factors. The results shown in **Figure 37**, include the component carryover factors K-W(B) (wing in presence of the body carryover due to angle of attack), K-B(W) (body in presence of the wing carryover due to angle of attack), KK-B(W) (body in presence of the wing carryover due to panel deflection), XCP-W(B) (wing in presence of the body carryover center of pressure), and XCP-B(W) (body in presence of the wing carryover center of pressure). This output is repeated for the body plus each additional aft fin set, if one exists on the configuration. If additional fin sets are present on the configuration additional pages are output with each one successively included.
- "COMPLETE CONFIGURATION" - Complete configuration aerodynamics. This output was illustrated in **Figure 21**. The values are obtained by summing the body-wing and tail in the presence of the wing flow field data.

If the PRINT AERO BEND or PART control card is used, the code will compute and print panel bending moment coefficients for each fin set on a separate page. One page is shown in **Figure 38**. The sign convention is that assumed for the individual panel loads and equivalent angles of attack, noted above. The bending moment coefficients are based upon the reference area and longitudinal length given at the top of the page. The moments are referenced about the fin-body structure specified by the root chord span station.

**Figure 39** illustrates the panel hinge moments coefficients computed when the control cards PRINT AERO HINGE or PART are used. The reference area and longitudinal reference length given at the top of the page are used. All moments are computed about the hinge line, which is defined using namelist DEFLCT.

If TRIM is specified, the user can selectively print the six untrimmed static aerodynamic tables used in the trim process to file for003.dat. An example is shown in **Figure 40**. The code computes the six-component aerodynamics at ten deflection angles for each specified angle of attack, then interpolates for  $C_m=0$ . Note that this trim process can be used to create control authority data, effectively giving the user 10 deflection angles, 20 angles of attack, and 20 Mach numbers per input case.

#### 4.2.3 Pressure Distribution Data

If the Mach number is supersonic ( $M \geq 1.2$ ), the user has the option to print the surface pressure distributions over the body and fins. This option is selected only through the addition of the control card PRESSURES. Since three body alone supersonic methods are available (Van Dyke Hybrid, Second-Order Shock Expansion (SOSE), and Newtonian flow) the capability exists to output the pressure distribution data from any one of these methods. The method to be used in the calculation of the pressure data is controlled with the control cards SOSE and HYPER; if neither control card is input, the Van Dyke Hybrid method is selected unless it is not valid for the case. Because of the nature of the calculations, body alone pressures are printed for angles of attack less than or equal to 15 degrees when using the Hybrid or SOSE techniques.

The primary body pressure distribution output is written to file "for010.dat". Local Mach number data is computed using the SOSE method only, and is written to file "for012.dat" if the PRESSURES option is used. All body pressure distribution data is based on a configuration that has body diameter of unity; that is, the configuration is expressed in calibers (or body diameters). The longitudinal stations at which pressure coefficient data is desired cannot be user specified; however, sufficient data is provided to permit accurate interpolation for most applications.

The capability also exists for the user to output the pressure distribution data over fins at any Mach number greater than 1.05. This option is also controlled by the PRESSURES control card. Due to the nature of the method, only pressure distribution data at zero angle of attack is presently output. The fin pressure data is written to file "for011.dat".

#### 4.3 DYNAMIC DERIVATIVES

As shown in **Figure 41**, the total configuration dynamic derivatives are provided in compact form for easy interpretation. The dynamic derivatives are summarized as a function of angle of attack in the user specified units. All derivatives are in the body axis system, with assumed rates of rotation also in that system. The coefficients provided are as follows:

CMQ	Pitching moment coefficient derivative with pitch rate: $C_{mq} = \partial C_m / \partial (q * l_{ref} / 2 / V)$
CNQ	Normal force coefficient derivative with pitch rate: $C_{Nq} = \partial C_N / \partial (q * l_{ref} / 2 / V)$
CAQ	Axial force coefficient derivative with pitch rate: $C_{Aq} = \partial C_A / \partial (q * l_{ref} / 2 / V)$
CMAD	Pitching moment derivative with rate of change of angle of attack: $C_{m\dot{\alpha}} = \partial C_m / \partial (\dot{\alpha} * l_{ref} / 2 / V)$

CNAD	Normal force derivative with rate of change of angle of attack: $C_{N\dot{\alpha}} = \partial C_N / \partial (\dot{\alpha} * l_{ref} / 2 / V)$
CLLP	Rolling moment coefficient derivative with roll rate: $C_{lp} = \partial C_l / \partial (p * l_{atref} / 2 / V)$
CLNP	Yawing moment coefficient derivative with roll rate: $C_{np} = \partial C_n / \partial (p * l_{atref} / 2 / V)$
CYP	Side force coefficient derivative with roll rate: $C_{Yp} = \partial C_Y / \partial (p * l_{atref} / 2 / V)$
CLLR	Rolling moment coefficient derivative with yaw rate: $C_{lr} = \partial C_l / \partial (r * l_{atref} / 2 / V)$
CLNR	Yawing moment coefficient derivative with yaw rate: $C_{nr} = \partial C_n / \partial (r * l_{atref} / 2 / V)$
CYR	Side force coefficient derivative with yaw rate: $C_{Yr} = \partial C_Y / \partial (r * l_{atref} / 2 / V)$

The dynamic derivatives are printed after all static coefficients and partial static aerodynamics are printed. If a BUILD or PART card is input, additional dynamic derivatives for partial configurations and/or configuration components are printed. The complete set of six force and moment components due to each of the three body axis rotations is written to the for042.dat file

#### 4.4 EXTERNAL DATA FILES

The code has the capability to be used in conjunction with other missile design tools, such as post-processing plotting programs or trajectory programs. Fixed format aerodynamic data is output as an external data file with the addition of the PLOT control card. The PLOT data are written to file “for003.dat”. Included in this data file are the six component forces and moments based upon the user specified reference quantities. In order to print component buildup data to the plot file the BUILD and PLOT control cards must be present in the case. If TRIM calculations were performed, the PLOT file also includes the control deflection for trim.

The output files “for020.dat” and “for021.dat” are generated if the control card PLOT is used. The file “for020.dat” generates data used by software developed with Adaptive Modeling Language (AML). The file “for021.dat” generates data that can be used with the Aviator Visual Design Simulator (AVDS). **Tables 25 and 26** list the contents of these files.

The output file “for042.csv” is always generated and contains the majority of the data supplied in the “for006.dat” file in a row/column file format with a header indicating the contents of the columns. **Table 27** lists the dictionary for the contents of the “for042.csv” file.

The output file “for043.csv” is generated if the control card PRINT AERO SYNTHS, PRINT AERO BEND, PRINT AERO HINGE, or PART is used. This file generates fin related variables in a comma delimited format. **Table 28** lists the contents of this file.

Table 25. Dictionary listing for the FOR020.DAT file

MACH	Mach Number
ALT	Altitude
VINF	Freestream Velocity
Q	Dynamic Pressure
ALPHA	Angle of Attack
BETA	Sideslip Angle
CN	Normal Force Coefficient
CM	Pitching Moment Coefficient
CA	Axial Force Coefficient
CY	Side-Force Coefficient
CLN	Yawing Moment Coefficient
CLL	Rolling Moment Coefficient
CL	Lift Force Coefficient
CD	Drag Force Coefficient
CL/CD	Lift-to-Drag Ratio
CNA	Normal Force Coefficient change wrt Angle of Attack
CMA	Pitching Moment Coefficient change wrt Angle of Attack
CYB	Side-force Coefficient change wrt Sideslip Angle
CLNB	Yawing Moment Coefficient change wrt Sideslip Angle
CLLB	Rolling Moment Coefficient change wrt Sideslip Angle
CMQ	Pitching Moment Coefficient change wrt Pitch Rate
CLNR	Yawing Moment Coefficient change wrt Yaw Rate
CLLP	Rolling Moment Coefficient change wrt Roll Rate

Table 26. Dictionary listing for the FOR021.DAT file

Row 1	
MACH	Mach Number
ALT	Altitude
BETA	Sideslip Angle
DELTA	Fin deflections
Row 2	
NALPHA	Number of angles of attack
NCOL	Number of columns of data in row 3
NCDD	Number of columns of dynamic derivatives in row 3
Row 3	
ALPHA	Angle of Attack
CN	Normal Force Coefficient
CM	Pitching Moment Coefficient
CA	Axial Force Coefficient
CY	Side-Force Coefficient
CLN	Yawing Moment Coefficient
CLL	Rolling Moment Coefficient
CNAD	Normal Force Coefficient change wrt Angle of Attack Change Rate
CMAD	Pitching Moment Coefficient change wrt Angle of Attack Change Rate
CNQ	Normal Force Coefficient change wrt Pitch Rate
CMQ	Pitching Moment Coefficient change wrt Pitch Rate
CAQ	Axial Force Coefficient change wrt Pitch Rate
CYQ	Side-force Coefficient change wrt Pitch Rate
CLNQ	Yawing Moment Coefficient change wrt Pitch Rate
CLLQ	Rolling Moment Coefficient change wrt Pitch Rate
CNP	Normal Force Coefficient change wrt Roll Rate
CMP	Pitching Moment Coefficient change wrt Roll Rate
CAP	Axial Force Coefficient change wrt Roll Rate
CYP	Side-force Coefficient change wrt Roll Rate
CLNP	Yawing Moment Coefficient change wrt Roll Rate
CLLP	Rolling Moment Coefficient change wrt Roll Rate
CNR	Normal Force Coefficient change wrt Yaw Rate
CMR	Pitching Moment Coefficient change wrt Yaw Rate
CAR	Axial Force Coefficient change wrt Yaw Rate
CYR	Side-force Coefficient change wrt Yaw Rate
CLNR	Yawing Moment Coefficient change wrt Yaw Rate
CLLR	Rolling Moment Coefficient change wrt Yaw Rate
Row 3 will be printed for each angle of attack in the case. Rows 1-3 are then repeated for each succeeding Mach number in the case. Damping derivatives will only be printed if the altitude is zero AND the sideslip angle is zero.	



Table 27. Dictionary listing for the FOR042.CSV file

CASE	Case Number currently being run
RAD?	Defines units as radians or degrees (0 = degrees, 1 = radians)
TRIM?	Defines case as trim case or not (0 = not trim, 1 = trim)
MACH	Mach Number
RE	Reynolds Number
ALT	Altitude
Q	Dynamic Pressure
BETA	Sideslip Angle
PHI	Roll Angle
SREF	Reference Area
XCG	Distance from nose to Center of Gravity
XMRP	Distance from nose to Center of Gravity measured in calibers
LREF	Longitudinal Reference Length
LATREF	Latitudinal Reference Length
ALPHA	Angle of Attack
CN	Normal Force Coefficient
CM	Pitching Moment Coefficient
CA	Axial Force Coefficient
CA_0B	Axial Force Coefficient with no base drag included
CA_FB	Axial Force Coefficient with full base drag included
CY	Side-Force Coefficient
CLN	Yawing Moment Coefficient
CLL	Rolling Moment Coefficient
CL	Lift Force Coefficient
CD	Drag Force Coefficient
CL/CD	Lift-to-Drag Ratio
X-C.P.	Distance from nose to Center of Pressure
CNA	Normal Force Coefficient change wrt Angle of Attack
CMA	Pitching Moment Coefficient change wrt Angle of Attack
CYB	Side-force Coefficient change wrt Sideslip Angle
CLNB	Yawing Moment Coefficient change wrt Sideslip Angle
CLLB	Rolling Moment Coefficient change wrt Sideslip Angle
CNQ	Normal Force Coefficient change wrt Pitch Rate
CMQ	Pitching Moment Coefficient change wrt Pitch Rate
CAQ	Axial Force Coefficient change wrt Pitch Rate
CNAD	Normal Force Coefficient change wrt Angle of Attack change Rate
CMAD	Pitching Moment Coefficient change wrt Angle of Attack change Rate
CYQ	Side-force Coefficient change wrt Pitch Rate
CLNQ	Yawing Moment Coefficient change wrt Pitch Rate
CLLQ	Rolling Moment Coefficient change wrt Pitch Rate
CYR	Side-force Coefficient change wrt Yaw Rate
CLNR	Yawing Moment Coefficient change wrt Yaw Rate
CLLR	Rolling Moment Coefficient change wrt Yaw Rate
CYP	Side-force Coefficient change wrt Roll Rate
CLNP	Yawing Moment Coefficient change wrt Roll Rate

Table 28. Dictionary listing for the FOR043.CSV file

CASE	Case Number Currently Being Run
MACH	Mach Number
ALPHA	Angle of Attack
CNx_IPB	Finset x Normal Force Coefficient in Presence of Body
CMx_IPB	Finset x Pitching Moment Coefficient in Presence of Body
CAX_IPB	Finset x Axial Force Coefficient in Presence of Body
CYx_IPB	Finset x Side Force Coefficient in Presence of Body
CLNx_IPB	Finset x Yawing moment Coefficient in Presence of Body
CLLx_IPB	Finset x Rolling Moment Coefficient in Presence of body
PBMx_Py	Fin y of Finset x Bending Moment Coefficient
PHMx_Py	Fin y of Finset x Hinge Moment Coefficient

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR
A - UNKNOWN VARIABLE NAME
B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME
C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)
D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED
E - ASSIGNED VALUES EXCEED ARRAY DIMENSION
F - SYNTAX ERROR

***** INPUT DATA CARDS *****

CASEID Missile Datcom ERROR flags                                comments

    NALPH = 4.0000,                                              << Mis-spelled
    ** ERROR ** 1*A 0*B 0*C 0*D 0*E 0*F
    * FATAL ERROR *

$REFQ SREF 100.,                                              << missing "="
    ** ERROR ** 0*A 1*B 0*C 0*D 0*E 0*F
    * FATAL ERROR *

    DNOSE(2) = 2.5,                                              << not an array
    ** ERROR ** 0*A 0*B 1*C 0*D 0*E 0*F
    * FATAL ERROR *

    NPANEL = 1., 2., 3.,                                        << not an array
    ** ERROR ** 0*A 0*B 0*C 1*D 0*E 0*F
    * FATAL ERROR *

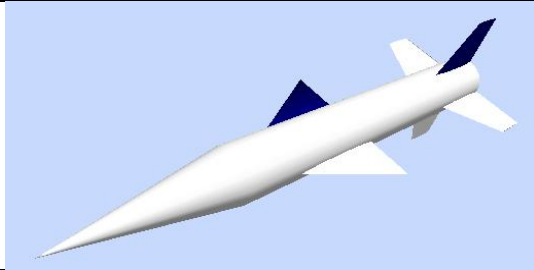
$FINSET1 PHIF(10) = 33.0, $                                    << exceeds array
    ** ERROR ** 0*A 0*B 0*C 0*D 1*E 0*F
    * FATAL ERROR *

$INLET NIN=1, $                                                << should be 1.
    ** ERROR ** 0*A 0*B 0*C 0*D 0*E 1*F

BUILT                                                         << Mis-spelled
    ** ERROR ** UNKNOWN CONTROL CARD - IGNORED
NEXT CASE
    FATAL ERROR ENCOUNTERED IN CONERR.
    PROGRAM STOPPED

```

Figure 18. Example of Missile Datcom Error Messages



```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****
                AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

CONERR - INPUT ERROR CHECKING

ERROR CODES - N* DENOTES THE NUMBER OF OCCURENCES OF EACH ERROR
A - UNKNOWN VARIABLE NAME
B - MISSING EQUAL SIGN FOLLOWING VARIABLE NAME
C - NON-ARRAY VARIABLE HAS AN ARRAY ELEMENT DESIGNATION - (N)
D - NON-ARRAY VARIABLE HAS MULTIPLE VALUES ASSIGNED
E - ASSIGNED VALUES EXCEED ARRAY DIMENSION
F - SYNTAX ERROR

***** INPUT DATA CARDS *****

1 CASEID  PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
2  $FLTCON  NALPHA=8.,NMACH=1.,MACH=2.36,REN=3000000.,
3           ALPHA=0.,4.,8.,12.,
4           ALPHA(5)=16.,20.,24.,28.,$
5  $REFQ  XCG=18.75,$
6  $AXIBOD  LNOSE=11.25,DNOSE=3.75,LCENTR=26.25,DEXIT=0.,$
7  $FINSET1  XLE=15.42,NPANEL=2.,PHIF=90.,270.,SWEEP=0.,STA=1.,
8           CHORD=6.96,0.,SSPAN=1.875,5.355,
9           ZUPPER=2*0.02238,LMAXU=0.238,0.238,
10          LFLATU=0.524,0.524,LER=2*0.015,$
11  $FINSET2  XLE=31.915,NPANEL=4.,PHIF=0.,90.,180.,270.,LER=2*0.015,
12          SWEEP=0.,STA=1.,SSPAN=1.875,6.26,CHORD=5.585,2.792,
13          ZUPPER=2*0.02238,LMAXU=2*0.288,LFLATU=2*0.428,$
14 PART
15 PLOT
16 DAMP
17 SOSE
18 SAVE
19 DIM IN
20 NEXT CASE
21 CASEID TRIM OF CASE NUMBER 1
22  $TRIM SET=2.,$
23 PRINT AERO TRIM
24 NEXT CASE

```

Figure 19. Case Input Listing

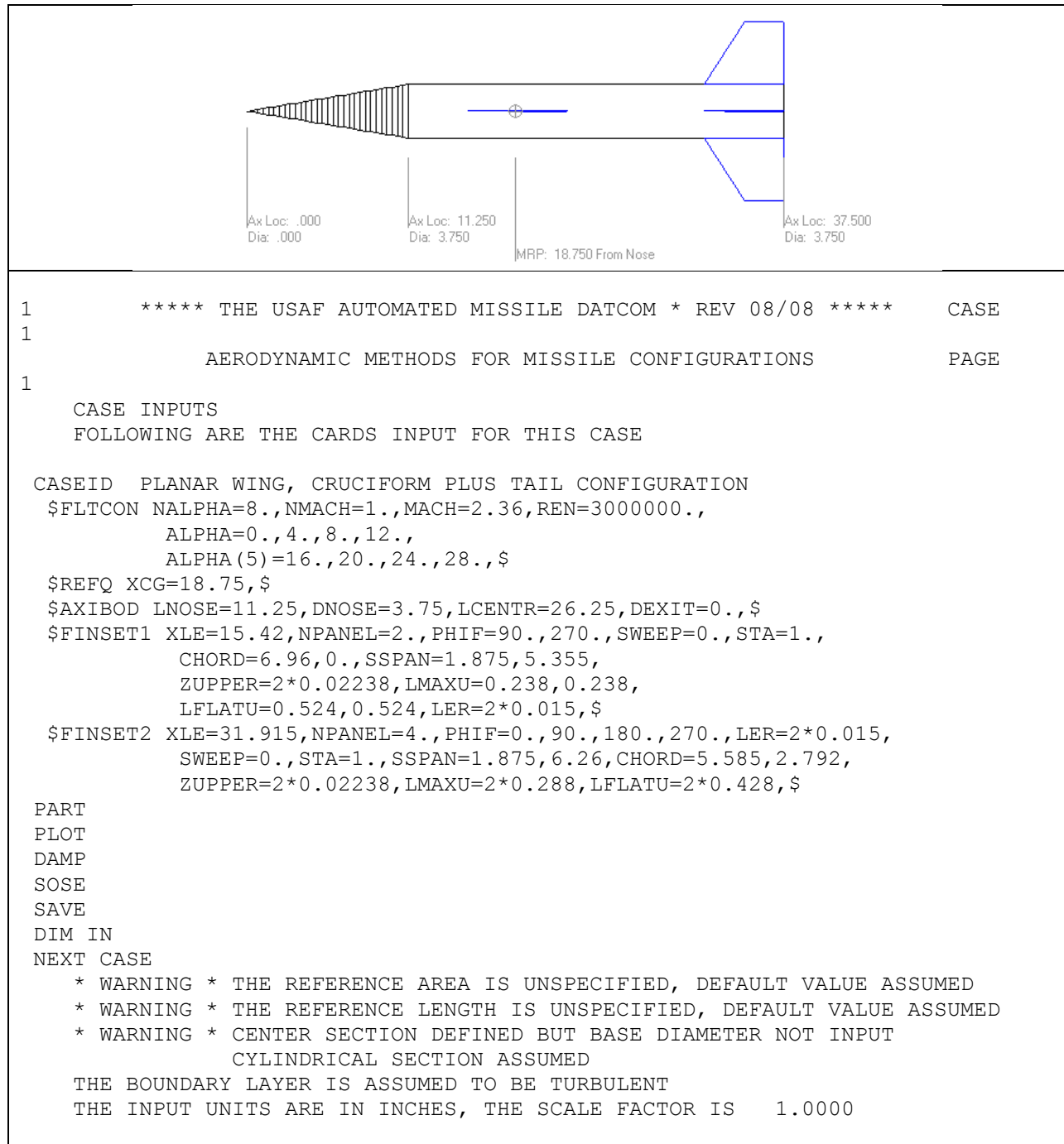


Figure 20. Example of Default Substitutions for Incomplete Case Inputs

1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****	CASE 1
	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS	PAGE 21
	PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION	
	STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2	
	***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****	
	MACH NO = 2.36 REYNOLDS NO = 3.000E+06 /FT	
	SIDESLIP = 0.00 DEG ROLL = 0.00 DEG	
	REF AREA = 11.045 IN**2 MOMENT CENTER = 18.750 IN	
	REF LENGTH = 3.75 IN LAT REF LENGTH = 3.75 IN	
	----- LONGITUDINAL ----- -- LATERAL DIRECTIONAL --	
	ALPHA CN CM CA CY CLN CLL	
	0.00 0.000 0.000 0.465 0.000 0.000 0.000	
	4.00 1.159 -1.429 0.464 0.000 0.000 0.000	
	8.00 2.476 -3.082 0.462 0.000 0.000 0.000	
	12.00 3.977 -4.833 0.458 0.000 0.000 0.000	
	16.00 5.554 -6.657 0.453 0.000 0.000 0.000	
	20.00 7.123 -8.576 0.447 0.000 0.000 0.000	
	24.00 8.573 -10.443 0.439 0.000 0.000 0.000	
	28.00 10.067 -12.345 0.430 0.000 0.000 0.000	
	ALPHA CL CD CL/CD X-C.P.	
	0.00 0.000 0.465 0.000 -1.220	
	4.00 1.124 0.544 2.068 -1.233	
	8.00 2.388 0.802 2.978 -1.245	
	12.00 3.795 1.275 2.977 -1.215	
	16.00 5.214 1.966 2.652 -1.198	
	20.00 6.541 2.856 2.290 -1.204	
	24.00 7.653 3.888 1.968 -1.218	
	28.00 8.686 5.106 1.701 -1.226	
	X-C.P. MEAS. FROM MOMENT CENTER IN REF. LENGTHS, NEG. AFT OF MOMENT CENTER	
1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****	CASE 1
	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS	PAGE 22
	PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION	
	STATIC AERODYNAMICS FOR BODY-FIN SET 1 AND 2	
	***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****	
	MACH NO = 2.36 REYNOLDS NO = 3.000E+06 /FT	
	SIDESLIP = 0.00 DEG ROLL = 0.00 DEG	
	REF AREA = 11.045 IN**2 MOMENT CENTER = 18.750 IN	
	REF LENGTH = 3.75 IN LAT REF LENGTH = 3.75 IN	
	----- DERIVATIVES (PER DEGREE) -----	
	ALPHA CNA CMA CYB CLNB CLLB	
	0.00 0.2702 -0.3297 -0.1994 0.4782 0.0000	
	4.00 0.3094 -0.3850 -0.2084 0.4692 -0.0156	
	8.00 0.3520 -0.4253 -0.2154 0.4406 -0.0232	
	12.00 0.3848 -0.4468 -0.2159 0.3541 -0.0162	
	16.00 0.3933 -0.4679 -0.2046 0.2233 0.0038	
	20.00 0.3773 -0.4733 -0.1899 0.1140 0.0297	
	24.00 0.3679 -0.4711 -0.1727 0.0369 0.0486	
	28.00 0.3790 -0.4798 -0.1613 -0.0167 0.0632	
	PANEL DEFLECTION ANGLES (DEGREES)	
	SET FIN 1 FIN 2 FIN 3 FIN 4 FIN 5 FIN 6 FIN 7 FIN 8	
	1 0.00 0.00	
	2 0.00 0.00 0.00 0.00	
	BODY ALONE LINEAR DATA GENERATED FROM SECOND ORDER SHOCK EXPANSION METHOD	

Figure 21. Total Configuration Aerodynamic Output Summary

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 2
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 2
              TRIM OF CASE NUMBER 1
              STATIC AERODYNAMIC COEFFICIENTS TRIMMED IN PITCH

              ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO = 2.36 REYNOLDS NO = 3.000E+06 /FT
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 11.045 IN**2 MOMENT CENTER = 18.750 IN
REF LENGTH = 3.75 IN LAT REF LENGTH = 3.75 IN

      ALPHA    DELTA    CL    CD    CN    CA
      0.00      0.00    0.000  0.465  0.000  0.465
      4.00     -3.48    0.794  0.524  0.829  0.467
      8.00     -7.33    1.685  0.696  1.765  0.455
     12.00    -11.13    2.712  1.021  2.864  0.435
     16.00    -14.74    3.755  1.506  4.025  0.413
     20.00    -18.60    4.707  2.141  5.155  0.402
     24.00    -22.58    5.487  2.867  6.179  0.387
     28.00     *NT*    *NT*    *NT*    *NT*    *NT*

PANELS FROM FIN SET 2 WERE DEFLECTED OVER THE RANGE -25.00 TO 20.00 DEG
PANEL 1 WAS FIXED
PANEL 2 WAS VARIED
PANEL 3 WAS FIXED
PANEL 4 WAS VARIED
NOTE - *NT* PRINTED WHEN NO TRIM POINT COULD BE FOUND
*** END OF JOB ***

```

Figure 22. Trimmed Output Summary

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 2
              PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
              AXISYMMETRIC BODY DEFINITION

              NOSE      CENTERBODY      AFT BODY      TOTAL
              OGIVE      CYLINDER      -----
SHAPE
LENGTH      11.250      26.250      0.000      37.500 IN
FINENESS RATIO      3.000      7.000      0.000      10.000
PLANFORM AREA      28.280      98.437      0.000      126.717 IN**2
AREA CENTROID      7.016      24.375      0.000      20.501 IN
WETTED AREA      89.818      309.251      0.000      399.069 IN**2
VOLUME      66.789      289.922      0.000      356.711 IN**3
VOL. CENTROID      7.714      24.375      0.000      21.255 IN

              MOLD LINE CONTOUR

LONGITUDINAL STATIONS      0.0000      1.1250      2.2500      3.3750      4.5000
5.6250      6.7500      7.8750      9.0000      10.1250      11.2500      13.8750
16.5000      19.1250      21.7500      24.3750      27.0000      29.6250      32.2500
34.8750      37.5000

              Z-OFFSET      0.0000      0.0000      0.0000      0.0000      0.0000
0.0000      0.0000      0.0000      0.0000      0.0000      0.0000      0.0000
0.0000      0.0000      0.0000      0.0000      0.0000      0.0000      0.0000
0.0000      0.0000

              BODY RADII      0.0000      0.3644      0.6871      0.9693      1.2119
1.4159      1.5819      1.7104      1.8019      1.8568      1.8750      1.8750
1.8750      1.8750      1.8750      1.8750      1.8750      1.8750      1.8750
1.8750      1.8750

```

Figure 23. Body Geometry Output



```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 3
              PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
              FIN SET NUMBER 1 AIRFOIL SECTION

              NACA S-3-23.8-04.5-52.4

              X/C      X-UPPER  Y-UPPER  X-LOWER  Y-LOWER  MEAN LINE  THICKNESS
0.00000  0.00000  0.00000  0.00000  0.00000  0.00000  0.00000
0.00100  0.00100  0.00009  0.00100 -0.00009  0.00000  0.00019
0.00200  0.00200  0.00019  0.00200 -0.00019  0.00000  0.00038
0.00300  0.00300  0.00028  0.00300 -0.00028  0.00000  0.00057
0.00400  0.00400  0.00038  0.00400 -0.00038  0.00000  0.00076
0.00500  0.00500  0.00047  0.00500 -0.00047  0.00000  0.00095
0.00600  0.00600  0.00057  0.00600 -0.00057  0.00000  0.00113
0.00800  0.00800  0.00076  0.00800 -0.00076  0.00000  0.00151
0.01000  0.01000  0.00095  0.01000 -0.00095  0.00000  0.00189
0.02000  0.02000  0.00189  0.02000 -0.00189  0.00000  0.00378
0.03000  0.03000  0.00284  0.03000 -0.00284  0.00000  0.00567
0.04000  0.04000  0.00378  0.04000 -0.00378  0.00000  0.00756
0.05000  0.05000  0.00473  0.05000 -0.00473  0.00000  0.00945
0.06000  0.06000  0.00567  0.06000 -0.00567  0.00000  0.01134
0.08000  0.08000  0.00756  0.08000 -0.00756  0.00000  0.01513
0.10000  0.10000  0.00945  0.10000 -0.00945  0.00000  0.01891
0.12000  0.12000  0.01134  0.12000 -0.01134  0.00000  0.02269
0.14000  0.14000  0.01324  0.14000 -0.01324  0.00000  0.02647
0.16000  0.16000  0.01513  0.16000 -0.01513  0.00000  0.03025
0.18000  0.18000  0.01702  0.18000 -0.01702  0.00000  0.03403
0.20000  0.20000  0.01891  0.20000 -0.01891  0.00000  0.03782
0.22000  0.22000  0.02080  0.22000 -0.02080  0.00000  0.04160
0.24000  0.24000  0.02250  0.24000 -0.02250  0.00000  0.04500
0.26000  0.26000  0.02250  0.26000 -0.02250  0.00000  0.04500
0.28000  0.28000  0.02250  0.28000 -0.02250  0.00000  0.04500
0.30000  0.30000  0.02250  0.30000 -0.02250  0.00000  0.04500
      :      :      :      :      :      :
      :      : (points deleted for editing purposes) :      :
      :      :      :      :      :      :
0.70000  0.70000  0.02250  0.70000 -0.02250  0.00000  0.04500
0.75000  0.75000  0.02250  0.75000 -0.02250  0.00000  0.04500
0.80000  0.80000  0.01891  0.80000 -0.01891  0.00000  0.03782
0.82000  0.82000  0.01702  0.82000 -0.01702  0.00000  0.03403
0.84000  0.84000  0.01513  0.84000 -0.01513  0.00000  0.03025
0.86000  0.86000  0.01324  0.86000 -0.01324  0.00000  0.02647
0.88000  0.88000  0.01134  0.88000 -0.01134  0.00000  0.02269
0.90000  0.90000  0.00945  0.90000 -0.00945  0.00000  0.01891
0.92000  0.92000  0.00756  0.92000 -0.00756  0.00000  0.01513
0.94000  0.94000  0.00567  0.94000 -0.00567  0.00000  0.01134
0.96000  0.96000  0.00378  0.96000 -0.00378  0.00000  0.00756
0.98000  0.98000  0.00189  0.98000 -0.00189  0.00000  0.00378
1.00000  1.00000  0.00000  1.00000  0.00000  0.00000  0.00000

```

Figure 24. Airfoil Geometry Output

1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****						CASE	1
	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS						PAGE	5
	PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION							
	GEOMETRIC RESULTS FOR FIN SETS							
	FIN SET NUMBER 1							
	(DATA FOR ONE PANEL ONLY)							
SEGMENT	PLAN	ASPECT	TAPER	L.E.	T.E.	M.A.C.	T/C	
NUMBER	AREA	RATIO	RATIO	SWEEP	SWEEP	CHORD	RATIO	
	IN**2			DEG	DEG	IN		
1	12.1104	1.000	0.000	63.435	0.000	4.640	0.045	
TOTAL	12.1104	1.000	0.000	63.435	0.000	4.640	0.045	
	FIN SET NUMBER 2							
	(DATA FOR ONE PANEL ONLY)							
SEGMENT	PLAN	ASPECT	TAPER	L.E.	T.E.	M.A.C.	T/C	
NUMBER	AREA	RATIO	RATIO	SWEEP	SWEEP	CHORD	RATIO	
	IN**2			DEG	DEG	IN		
1	18.3666	1.047	0.500	32.495	0.000	4.344	0.045	
TOTAL	18.3666	1.047	0.500	32.495	0.000	4.344	0.045	

Figure 25. Fin Geometry Output

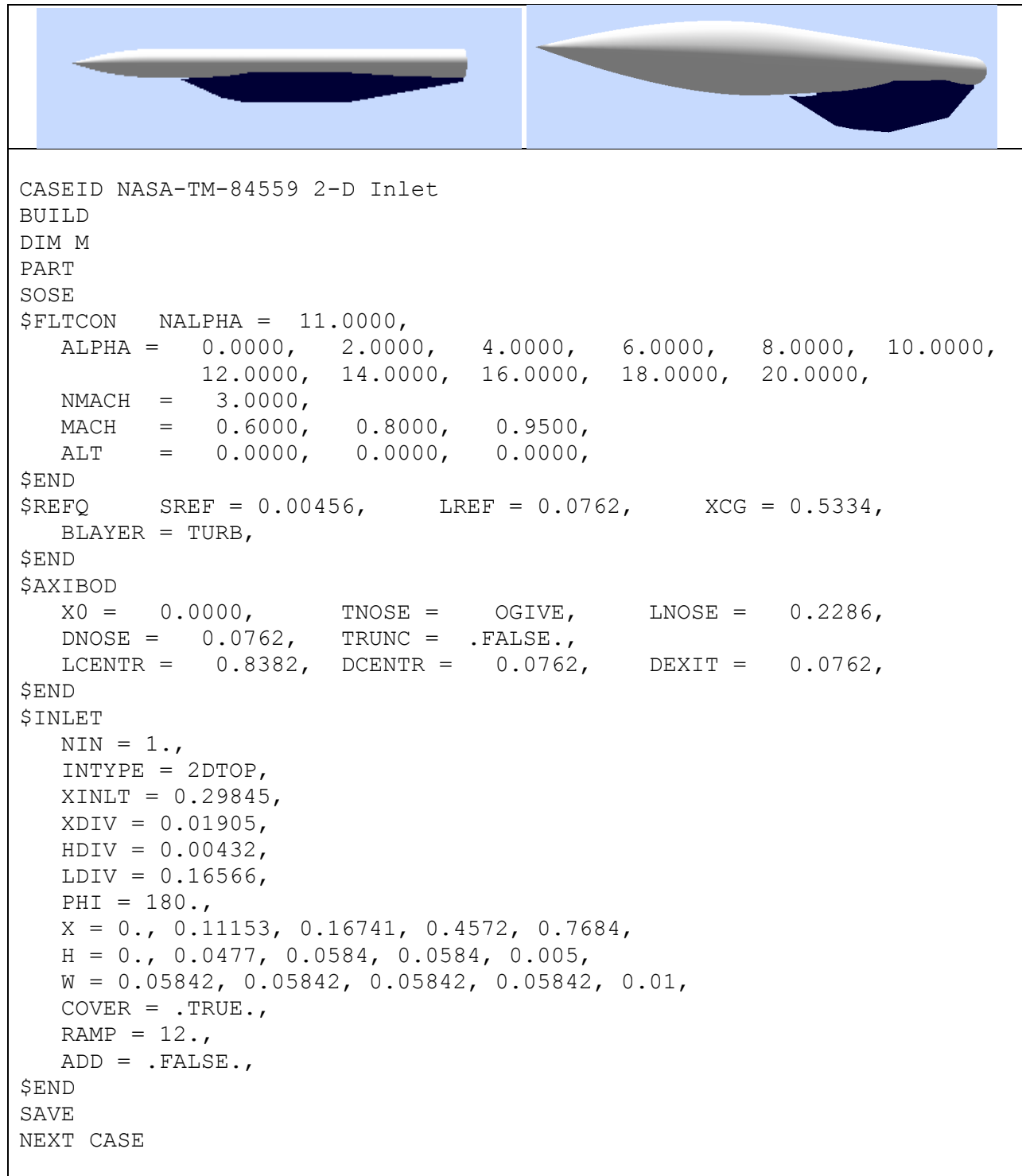


Figure 26. Inlet Geometry Input

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 3
              NASA-TM-84559 2-D Inlet
              INLET GEOMETRY

INLET IS A TOP MOUNTED TWO-DIMENSIONAL TYPE

THE INLETS ARE COVERED

NUMBER OF INLETS = 1

INLET ANGULAR ROLL POSITIONS FROM TOP VERTICAL CENTER (DEG)
    (SAME CONVENTION AS FIN ROLL POSITIONS)
    180.0

LONGITUDINAL DISTANCE FROM MISSILE NOSE TIP TO
INLET LEADING EDGE = 0.30

      INLET POSITIONS RELATIVE TO THE LEADING EDGE
      POSITION              LONGITUDINAL      WIDTH      HEIGHT
TOP LIP LEADING EDGE      0.000      0.058      0.000
COWL LIP LEADING EDGE     0.112      0.058      0.048
MID BODY START            0.167      0.058      0.058
BOATTAIL START            0.457      0.058      0.058
BOATTAIL END              0.768      0.010      0.005

LONGITUDINAL DISTANCE FROM INLET LEADING EDGE TO
DIVERTER LEADING EDGE = 0.02

DIVERTER LENGTH = 0.17

HEIGHT OF DIVERTER LEADING EDGE = 0.00

```

Figure 27. Inlet Geometry Output

```

1      ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****      CASE      1
      AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS      PAGE      6
      BASE-JET PLUME INTERACTION FLOW PARAMETERS

      ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO   =          2.36              REYNOLDS NO = 3.000E+06 /FT
SIDESLIP  =          0.00 DEG              ROLL   =          0.00 DEG
REF AREA  =        11.045 FT**2          MOMENT CENTER =        18.750 FT
REF LENGTH =          3.75 FT          LAT REF LENGTH =          3.75 FT

      WARNING EXTRAPOLATION WILL BE REQUIRED FOR THE FOLLOWING CONDITIONS:

      * ANGLE OF ATTACK GREATER THAN 8.0

      ----- BASE FLOW PARAMETERS -----      ----- INCREMENTAL DATA -----
ALPHA      CP-BASE  CA-BASE  TB/TINF  PB/PINF      DEL CN      DEL CM      DEL CA
0.00      0.0828  -0.0162  5.9018  1.3226      0.0000      0.0000     -0.0010
4.00      0.0828  -0.0162  5.9018  1.3226      0.0004     -0.0028     -0.0010
8.00      0.0828  -0.0162  5.9018  1.3226      0.0009     -0.0057     -0.0010
12.00     0.0828  -0.0162  5.9018  1.3226      0.0013     -0.0085     -0.0010
16.00     0.0828  -0.0162  5.9018  1.3226      0.0017     -0.0113     -0.0010
20.00     0.0828  -0.0162  5.9018  1.3226      0.0022     -0.0142     -0.0010
24.00     0.0828  -0.0162  5.9018  1.3226      0.0026     -0.0170     -0.0010
28.00     0.0828  -0.0162  5.9018  1.3226      0.0030     -0.0198     -0.0010

```

Figure 28. Base-Jet Plume Interaction Output

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 6
              Protuberance Example Case
              PROTUBERANCE OUTPUT

          ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO =      0.40                      REYNOLDS NO = 3.000E+06 /FT
SIDESLIP =      0.00 DEG                  ROLL =      0.00 DEG
REF AREA =     113.097 IN**2              MOMENT CENTER = 39.000 IN
REF LENGTH =    12.00 IN                  LAT REF LENGTH = 12.00 IN

PROTUBERANCE AXIAL FORCE COEFFICIENT IS CALCULATED AT ZERO ANGLE OF
ATTACK AND ASSUMED CONSTANT FOR ALL ANGLES OF ATTACK. PROTUBERANCES
ARE CONSIDERED PART OF THE BODY WHEN CALCULATING TOTAL AXIAL FORCE.
PROTUBERANCE AXIAL FORCE IS INCLUDED IN THE TOTAL CONFIGURATION RESULTS.

----- PROTUBERANCE CALCULATIONS (AT ZERO ANGLE OF ATTACK) -----

NUMBER          TYPE          LONG.    NUMBER  INDIVIDUAL    TOTAL
                LOCATION (IN)              CA          CA
1              FAIRING          14.000      4      0.0009      0.0037

    **** INDUCED MOMENT BREAKDOWN ****
    PROTUBERANCE PHI = 0.000 CM = 0.0005 CLN = 0.0000
    PROTUBERANCE PHI = 90.000 CM = 0.0000 CLN = 0.0005
    PROTUBERANCE PHI = 180.000 CM = -0.0005 CLN = 0.0000
    PROTUBERANCE PHI = 270.000 CM = 0.0000 CLN = -0.0005

2      VERTICAL CYLINDER      22.000      4      0.0018      0.0073

    **** INDUCED MOMENT BREAKDOWN ****
    PROTUBERANCE PHI = 45.000 CM = 0.0007 CLN = 0.0007
    PROTUBERANCE PHI = 135.000 CM = -0.0007 CLN = 0.0007
    PROTUBERANCE PHI = 225.000 CM = -0.0007 CLN = -0.0007
    PROTUBERANCE PHI = 315.000 CM = 0.0007 CLN = -0.0007

3      LAUNCH SHOE          39.000      1      0.0028      0.0028
    **** INDUCED MOMENT BREAKDOWN ****
    PROTUBERANCE PHI = 0.000 CM = 0.0014 CLN = 0.0000

4      FLAT PLATE OR BLOCK    56.000      1      0.0298      0.0298
    **** INDUCED MOMENT BREAKDOWN ****
    PROTUBERANCE PHI = 0.000 CM = 0.0167 CLN = 0.0000

TOTAL CA DUE TO PROTUBERANCES = 0.0435

TOTAL CM DUE TO PROTUBERANCES = 0.0182

TOTAL CLN DUE TO PROTUBERANCES = 0.0000

```

Figure 29. Protuberance Output

1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****					CASE	1		
	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS					PAGE	6		
	PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION								
	BODY ALONE PARTIAL OUTPUT								
	***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****								
	MACH NO =	2.36		REYNOLDS NO =	3.000E+06 /FT				
	SIDESLIP =	0.00 DEG		ROLL =	0.00 DEG				
	REF AREA =	11.045 IN**2		MOMENT CENTER =	18.750 IN				
	REF LENGTH =	3.75 IN		LAT REF LENGTH =	3.75 IN				
	ALPHA	CA-FRIC	CA-PRES/WAVE	CA-BASE	CA-PROT	CA-SEP	CA-ALP		
	0.00	0.0845	0.1029	0.1254			0.0000		
	4.00	0.0841	0.1029	0.1251			0.0000		
	8.00	0.0828	0.1029	0.1242			0.0000		
	12.00	0.0808	0.1028	0.1227			0.0000		
	16.00	0.0781	0.1026	0.1205			0.0000		
	20.00	0.0746	0.1025	0.1178			0.0000		
	24.00	0.0705	0.1023	0.1146			0.0000		
	28.00	0.0659	0.1020	0.1107			0.0000		
	CROSS FLOW DRAG PROPORTIONALITY FACTOR = 1.00000								
	ALPHA	CN-POTEN	CN-VISC	CN-SEP	CM-POTEN	CM-VISC	CM-SEP	CM-PROT	CDC
	0.00	0.000	0.000		0.000	0.000			0.740
	4.00	0.222	0.047		0.582	-0.022			0.841
	8.00	0.438	0.232		1.151	-0.108			1.044
	12.00	0.645	0.665		1.693	-0.310			1.340
	16.00	0.836	1.308		2.196	-0.610			1.500
	20.00	1.009	2.005		2.649	-0.936			1.494
	24.00	1.159	2.608		3.042	-1.218			1.374
	28.00	1.282	3.330		3.367	-1.555			1.317

Figure 30. Body Alone Aerodynamic Partial Output

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
          AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 10
          PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
          FIN SET 2 CN, CM PARTIAL OUTPUT

          ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO = 2.36 REYNOLDS NO = 3.000E+06 /FT
SIDESLIP = 0.00 DEG ROLL = 0.00 DEG
REF AREA = 11.045 IN**2 MOMENT CENTER = 18.750 IN
REF LENGTH = 3.75 IN LAT REF LENGTH = 3.75 IN

NORMAL FORCE SLOPE AT ALPHA ZERO, CNA = 0.04956/DEG (1 PANEL)
FIN AERODYNAMIC CENTER = 0.53282 (FRACTION OF ROOT CHORD)
FIN PLANFORM CENTROID = 0.61113 (FRACTION OF ROOT CHORD)

ALPHA      CN      CN      CN      CM      CM      CM
          LINEAR  NON-LINEAR  TOTAL  LINEAR  NON-LINEAR  TOTAL
0.00      0.0000   0.0000   0.0000   0.0000   0.0000   0.0000
4.00      0.3952   0.0033   0.3985  -1.7043  -0.0143  -1.7186
8.00      0.7826   0.0264   0.8091  -3.3820  -0.1143  -3.4962
12.00     1.1549   0.0900   1.2449  -4.9996  -0.3897  -5.3893
16.00     1.5047   0.2185   1.7232  -6.5251  -0.9477  -7.4727
20.00     1.8251   0.3375   2.1626  -7.9279  -1.4658  -9.3937
24.00     2.1101   0.4759   2.5860  -9.1799  -2.0706  -11.2505
28.00     2.3540   0.6323   2.9863 -10.2561 -2.7551 -13.0112

```

Figure 31. Fin Normal Force and Pitching Moment Partial Output



```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 9
              PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
              FIN SET 2 CA PARTIAL OUTPUT

          ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO =      2.36 REYNOLDS NO = 3.000E+06 /FT
SIDESLIP =      0.00 DEG ROLL =      0.00 DEG
REF AREA =     11.045 IN**2 MOMENT CENTER =     18.750 IN
REF LENGTH =      3.75 IN LAT REF LENGTH =      3.75 IN

SINGLE FIN PANEL ZERO-LIFT AXIAL FORCE COMPONENTS

SKIN FRICTION      0.0116
SUBSONIC PRESSURE   0.0000
TRANSONIC WAVE      0.0000
SUPERSONIC WAVE     0.0092
LEADING EDGE        0.0096
TRAILING EDGE       0.0000
TOTAL CAO           0.0304

FIN AXIAL FORCE DUE TO ANGLE OF ATTACK

ALPHA      CA DUE TO LIFT (SINGLE PANEL)      CA-TOTAL (4 FINS)

    0.00      0.0000      0.1216
    4.00      0.0000      0.1214
    8.00      0.0000      0.1205
   12.00      0.0000      0.1190
   16.00      0.0000      0.1169
   20.00      0.0000      0.1143
   24.00      0.0000      0.1111
   28.00      0.0000      0.1074

```

Figure 32. Fin Axial Force Partial Output

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 5
              NACA 4312 AIRFOIL SECTION CHECK
              FIN SET 1 SECTION AERODYNAMICS

              IDEAL ANGLE OF ATTACK = 1.1260 DEG.
              ZERO LIFT ANGLE OF ATTACK = -3.3053 DEG.
              IDEAL LIFT COEFFICIENT = 0.5441
              ZERO LIFT PITCHING MOMENT COEFFICIENT = -0.0895
              MACH ZERO LIFT-CURVE-SLOPE = 0.1020 /DEG.
              LEADING EDGE RADIUS = 0.0159 FRACTION CHORD
              MAXIMUM AIRFOIL THICKNESS = 0.1200 FRACTION CHORD
              DELTA-Y = 3.1690 PERCENT CHORD

MACH = 0.300 CL-ALPHA = 0.1084 /DEG. XAC = 0.2484 CL MAX = 1.5493
MACH = 0.500 CL-ALPHA = 0.1158 /DEG. XAC = 0.2620 CL MAX = 1.5874
MACH = 0.700 CL-ALPHA = 0.1282 /DEG. XAC = 0.2879 CL MAX = 1.6254

*** CREST CRITICAL MACH NUMBER EXCEEDED ***

```

Figure 33. Airfoil Section Aerodynamic Partial Output

1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****	CASE	1
	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS	PAGE	5
	NASA-TM-84559 2-D Inlet		
	INLET AERODYNAMIC INCREMENTALS		
	***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****		
	MACH NO = 0.60	REYNOLDS NO = 1.391E+07 /M	
	ALTITUDE = 0.0 M	DYNAMIC PRESSURE = 25533.94 N/M**2	
	SIDESLIP = 0.00 DEG	ROLL = 0.00 DEG	
	REF AREA = 0.005 M**2	MOMENT CENTER = 0.533 M	
	REF LENGTH = 0.08 M	LAT REF LENGTH = 0.08 M	
	ALPHA	CN-INLT	CM-INLT
	0.00	0.0000	-0.0806
	2.00	0.0676	0.0611
	4.00	0.1458	0.1999
	6.00	0.2342	0.3350
	8.00	0.3324	0.4659
	10.00	0.4398	0.5918
	12.00	0.5560	0.7121
	14.00	0.6804	0.8264
	16.00	0.8124	0.9339
	18.00	0.9514	1.0343
	20.00	1.0971	1.1268
	CA-INLT	CA-ADD	CY-INLT
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	0.0914		0.0000
	CLN-INLT	CLL-INLT	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	
	0.0000	0.0000	

Figure 34. Inlet Aerodynamic Partial Output

1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****					CASE	1
	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS					PAGE	12
	PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION						
	AERODYNAMIC FORCE AND MOMENT SYNTHESIS						
	***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****						
	MACH NO =	2.36		REYNOLDS NO =	3.000E+06 /FT		
	SIDESLIP =	0.00 DEG		ROLL =	0.00 DEG		
	REF AREA =	11.045 IN**2		MOMENT CENTER =	18.750 IN		
	REF LENGTH =	3.75 IN		LAT REF LENGTH =	3.75 IN		
	-----FIN SET 2 IN PRESENCE OF THE BODY-----						
	ALPHA	CN	CM	CA	CY	CLN	CLL
	0.00	0.0000	0.0000	0.1216	0.0000	0.0000	0.0000
	4.00	0.3746	-1.6154	0.1216	0.0000	0.0000	0.0000
	8.00	0.7747	-3.3470	0.1216	0.0000	0.0000	0.0000
	12.00	1.1636	-5.0356	0.1216	0.0000	0.0000	0.0000
	16.00	1.5457	-6.6987	0.1216	0.0000	0.0000	0.0000
	20.00	1.9335	-8.3913	0.1216	0.0000	0.0000	0.0000
	24.00	2.3027	-10.0073	0.1216	0.0000	0.0000	0.0000
	28.00	2.6555	-11.5556	0.1216	0.0000	0.0000	0.0000

Figure 35. Fin Set in Presence of the Body Partial Output

```

1          ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE 1
              AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE 14
              PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
              AERODYNAMIC FORCE AND MOMENT SYNTHESIS

              ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO =      2.36 REYNOLDS NO = 3.000E+06 /FT
SIDESLIP =      0.00 DEG ROLL =      0.00 DEG
REF AREA =     11.045 IN**2 MOMENT CENTER = 18.750 IN
REF LENGTH =    3.75 IN LAT REF LENGTH = 3.75 IN

              -----FIN SET 2 PANEL CHARACTERISTICS-----

ALPHA   PANEL   AEQ(PANEL AXIS SYS.)   PANEL_CN   PANEL_XCP/CR   PANEL_YCP/(B/2)

0.00    1       0.0000                0.0000      0.5328         0.4219
0.00    2       0.0000                0.0000      0.5328         0.4219
0.00    3       0.0000                0.0000      0.5328         0.4219
0.00    4       0.0000                0.0000      0.5328         0.4219

4.00    1       0.0000                0.0000      0.5328         0.4219
4.00    2       3.7624                0.1873      0.5383         0.4232
4.00    3       0.0000                0.0000      0.5328         0.4219
4.00    4       -3.7624                -0.1873     0.5383         0.4232

8.00    1       0.0000                0.0000      0.5328         0.4219
8.00    2       7.6721                0.3873      0.5438         0.4245
8.00    3       0.0000                0.0000      0.5328         0.4219
8.00    4       -7.6721                -0.3873     0.5438         0.4245

12.00   1       0.0000                0.0000      0.5328         0.4219
12.00   2       11.2753               0.5818      0.5486         0.4259
12.00   3       0.0000                0.0000      0.5328         0.4219
12.00   4       -11.2753              -0.5818     0.5486         0.4259

16.00   1       0.0000                0.0000      0.5328         0.4219
16.00   2       14.5505               0.7728      0.5528         0.4272
16.00   3       0.0000                0.0000      0.5328         0.4219
16.00   4       -14.5505              -0.7728     0.5528         0.4272

20.00   1       0.0000                0.0000      0.5328         0.4219
20.00   2       17.8953               0.9667      0.5569         0.4285
20.00   3       0.0000                0.0000      0.5328         0.4219
20.00   4       -17.8953              -0.9667     0.5569         0.4285

24.00   1       0.0000                0.0000      0.5328         0.4219
24.00   2       21.3032               1.1513      0.5609         0.4297
24.00   3       0.0000                0.0000      0.5328         0.4219
24.00   4       -21.3032              -1.1513     0.5609         0.4297

28.00   1       0.0000                0.0000      0.5328         0.4219
28.00   2       24.6754               1.3277      0.5646         0.4309
28.00   3       0.0000                0.0000      0.5328         0.4219
28.00   4       -24.6754              -1.3277     0.5646         0.4309

```

Figure 36. Fin Set in Presence of the Body Partial Output (continued)

1

\*\*\*\*\* THE USAF AUTOMATED MISSILE DATCOM \* REV 08/08 \*\*\*\*\*

CASE 1

AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS

PAGE 15

PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION

AERODYNAMIC FORCE AND MOMENT SYNTHESIS

\*\*\*\*\* FLIGHT CONDITIONS AND REFERENCE QUANTITIES \*\*\*\*\*

MACH NO = 2.36 REYNOLDS NO = 3.000E+06 /FT

SIDESLIP = 0.00 DEG ROLL = 0.00 DEG

REF AREA = 11.045 IN\*\*2 MOMENT CENTER = 18.750 IN

REF LENGTH = 3.75 IN LAT REF LENGTH = 3.75 IN

CARRYOVER INTERFERENCE FACTORS - FIN SET 1

ALPHA	K-W (B)	K-B (W)	KK-W (B)	KK-B (W)	XCP-B (W)
0.00	1.4031	0.4360	0.9347	0.3658	1.0903
4.00	1.3650	0.4360	0.9347	0.3658	1.0903
8.00	1.3042	0.4360	0.9347	0.3658	1.0903
12.00	1.2404	0.4360	0.9347	0.3658	1.0903
16.00	1.1819	0.4360	0.9347	0.3658	1.0903
20.00	1.1325	0.4360	0.9347	0.3658	1.0903
24.00	1.0934	0.4360	0.9347	0.3658	1.0903
28.00	1.0638	0.4360	0.9347	0.3658	1.0903

Figure 37. Carryover Interference Factors Partial Output

1 \*\*\*\*\* THE USAF AUTOMATED MISSILE DATCOM \* REV 08/08 \*\*\*\*\* CASE

1

18 AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE

PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION

FIN SET 2 PANEL BENDING MOMENTS (ABOUT EXPOSED ROOT CHORD)

\*\*\*\*\* FLIGHT CONDITIONS AND REFERENCE QUANTITIES \*\*\*\*\*

MACH NO = 2.36 REYNOLDS NO = 3.000E+06 /FT

SIDESLIP = 0.00 DEG ROLL = 0.00 DEG

REF AREA = 11.045 IN\*\*2 MOMENT CENTER = 18.750 IN

REF LENGTH = 3.75 IN LAT REF LENGTH = 3.75 IN

ALPHA PANL 1 PANL 2 PANL 3 PANL 4 PANL 5 PANL 6 PANL 7 PANL

8

0.0 0.00E+00 0.00E+00 0.00E+00 0.00E+00

4.0 2.21E-08 1.11E-01 -9.30E-09 -1.11E-01

8.0 4.21E-08 2.23E-01 -2.33E-08 -2.23E-01

12.0 8.01E-08 3.27E-01 -5.81E-08 -3.27E-01

16.0 8.10E-08 4.26E-01 -3.83E-08 -4.26E-01

20.0 7.79E-08 5.16E-01 -1.29E-07 -5.16E-01

24.0 -1.16E-08 6.07E-01 -6.45E-08 -6.07E-01

28.0 2.50E-08 6.92E-01 -7.92E-08 -6.92E-01

Figure 38. Panel Bending Moment Partial Output

1	***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 *****							CASE
1	AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS							PAGE
20	PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION							
	FIN SET 2 PANEL HINGE MOMENTS (ABOUT HINGE LINE)							
	***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****							
	MACH NO =	2.36			REYNOLDS NO =	3.000E+06 /FT		
	SIDESLIP =	0.00 DEG			ROLL =	0.00 DEG		
	REF AREA =	11.045 IN**2			MOMENT CENTER =	18.750 IN		
	REF LENGTH =	3.75 IN			LAT REF LENGTH =	3.75 IN		
	ALPHA	PANL 1	PANL 2	PANL 3	PANL 4	PANL 5	PANL 6	PANL 7
8								
	0.0	0.00E+00	0.00E+00	0.00E+00	0.00E+00			
	4.0	-2.19E-09	-1.32E-02	9.21E-10	1.32E-02			
	8.0	-4.17E-09	-3.03E-02	2.31E-09	3.03E-02			
	12.0	-7.94E-09	-4.91E-02	5.75E-09	4.91E-02			
	16.0	-8.03E-09	-6.88E-02	3.80E-09	6.88E-02			
	20.0	-7.72E-09	-8.93E-02	1.27E-08	8.93E-02			
	24.0	1.15E-09	-1.11E-01	6.39E-09	1.11E-01			
	28.0	-2.47E-09	-1.34E-01	7.85E-09	1.34E-01			

Figure 39. Panel Hinge Moment Partial Output

VARIABLES=ALPHA,CN,CM,CA,CY,CLN,CLL,DELTA,CL,CD							
ZONE T="UNTRIMMED MACH= 0.30"							
-3.00	-6.8754	18.8254	0.4952	0.0000	0.0000	0.0000	-25.0000
-2.00	-5.6970	15.5945	0.5283	0.0000	0.0000	0.0000	-25.0000
-1.00	-4.5390	12.5304	0.5438	0.0000	0.0000	0.0000	-25.0000
0.00	-3.4086	9.4947	0.5420	0.0000	0.0014	0.0140	-25.0000
1.00	-2.2997	6.4886	0.5226	0.0000	0.0033	0.0253	-25.0000
2.00	-1.2134	3.5370	0.4864	0.0000	0.0084	0.0411	-25.0000
3.00	-0.1484	0.6385	0.4333	0.0000	0.0365	0.0532	-25.0000
ZONE T="UNTRIMMED MACH= 0.30"							
-3.00	-6.2225	16.7872	0.4487	0.0000	0.0000	0.0000	-20.0000
-2.00	-5.0514	13.5915	0.4841	0.0000	0.0000	0.0000	-20.0000
-1.00	-3.9059	10.5543	0.5021	0.0000	0.0000	0.0000	-20.0000
0.00	-2.7851	7.5369	0.5023	0.0000	0.0013	0.0132	-20.0000
1.00	-1.6890	4.5598	0.4850	0.0000	0.0035	0.0230	-20.0000
2.00	-0.6164	1.6410	0.4507	0.0000	0.0118	0.0374	-20.0000
3.00	0.4450	-1.2566	0.3982	0.0000	-0.0270	0.0484	-20.0000
ZONE T="UNTRIMMED MACH= 0.30"							
-3.00	-5.1144	13.3298	0.3892	0.0000	0.0000	0.0000	-15.0000
-2.00	-3.9649	10.2180	0.4286	0.0000	0.0000	0.0000	-15.0000
-1.00	-2.8370	7.2158	0.4499	0.0000	0.0000	0.0000	-15.0000
0.00	-1.7371	4.2442	0.4532	0.0000	0.0015	0.0115	-15.0000
1.00	-0.6660	1.3268	0.4388	0.0000	0.0058	0.0196	-15.0000
2.00	0.3961	-1.5770	0.4058	0.0000	-0.0172	0.0309	-15.0000
3.00	1.4661	-4.5196	0.3526	0.0000	-0.0034	0.0399	-15.0000

Figure 40. Untrimmed Aerodynamic Output ("for003.dat")

```

1      ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE
1
25     AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE

        PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
        BODY + 2 FIN SETS DYNAMIC DERIVATIVES

        ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO =      2.36                      REYNOLDS NO = 3.000E+06 /FT
SIDESLIP =      0.00 DEG                      ROLL =      0.00 DEG
REF AREA =      11.045 IN**2                MOMENT CENTER =      18.750 IN
REF LENGTH =      3.75 IN                    LAT REF LENGTH =      3.75 IN

        ----- DYNAMIC DERIVATIVES (PER DEGREE) -----
        ALPHA      CNQ      CMQ      CAQ      CNAD      CMAD
          0.00      1.854      -7.598      0.000      1.238      -1.923
          4.00      1.877      -7.697      0.000      1.238      -1.923
          8.00      1.818      -7.466      0.000      1.238      -1.923
         12.00      1.723      -7.089      0.000      1.238      -1.923
         16.00      1.600      -6.588      0.000      1.238      -1.923
         20.00      1.505      -6.202      0.000      1.238      -1.923
         24.00      1.410      -5.817      0.000      1.238      -1.923
         28.00      1.318      -5.440      0.000      1.238      -1.923

        PITCH RATE DERIVATIVES NON-DIMENSIONALIZED BY Q*LREF/2*V
1      ***** THE USAF AUTOMATED MISSILE DATCOM * REV 08/08 ***** CASE
1
26     AERODYNAMIC METHODS FOR MISSILE CONFIGURATIONS PAGE

        PLANAR WING, CRUCIFORM PLUS TAIL CONFIGURATION
        BODY + 2 FIN SETS DYNAMIC DERIVATIVES

        ***** FLIGHT CONDITIONS AND REFERENCE QUANTITIES *****
MACH NO =      2.36                      REYNOLDS NO = 3.000E+06 /FT
SIDESLIP =      0.00 DEG                      ROLL =      0.00 DEG
REF AREA =      11.045 IN**2                MOMENT CENTER =      18.750 IN
REF LENGTH =      3.75 IN                    LAT REF LENGTH =      3.75 IN

        ----- DYNAMIC DERIVATIVES (PER DEGREE) -----
        ALPHA      CYR      CLNR      CLLR      CYP      CLNP      CLLP
          0.00      1.843      -7.804      0.000      0.000      0.000      -0.466
          4.00      1.811      -7.663      0.000      0.002      -0.009      -0.469
          8.00      1.765      -7.464      0.000      0.004      -0.019      -0.475
         12.00      1.724      -7.284      0.000      0.006      -0.028      -0.483
         16.00      1.694      -7.156      0.000      0.008      -0.034      -0.476
         20.00      1.679      -7.092      0.000      0.009      -0.038      -0.448
         24.00      1.680      -7.096      0.000      0.009      -0.038      -0.429
         28.00      1.697      -7.169      0.000      0.008      -0.035      -0.408

        YAW AND ROLL RATE DERIVATIVES NON-DIMENSIONALIZED BY R*LATREF/2*V

```

Figure 41. Dynamic Derivative Output



## 5.0 AERODYNAMIC METHODOLOGY

The methods incorporated in the program are summarized at the end of this section. Most methods are coded in individual subroutines so that revision or replacement is easily accomplished. The program subroutines corresponding to the individual methods are given in tables immediately following the tables of methods. The tables are listed in the following order:

- Table 29. Body Alone Aerodynamic Methodology
- Table 30. Body Alone Subroutines
- Table 31. Fin Alone Aerodynamic Methodology
- Table 32. Fin Alone Subroutines
- Table 33. Inlet Aerodynamic Methodology
- Table 34. Inlet Subroutines
- Table 35. Body-Fin Synthesis Aerodynamic Methodology
- Table 36. Body-Fin Synthesis Subroutines

Table 29. Body Alone Aerodynamic Methodology

Parameter	Subsonic/Transonic ( $M < 1.2$ )	Supersonic ( $M > 1.2$ )
CN-potential	Option 1: Nose-cylinder: MBB charts, TN-WE-2-9769 Boattail: NSWC charts, NSWC-TR-81-156 Flare: Army charts AMCP 706-280 Option 2: Slender Body Theory Camber: AIAA 2003-3668 (Option 2 only)	Option 1 and Option 2: Second Order Shock Expansion, NSWC-TR-81-156 Van Dyke Hybrid theory, NSWC-TR-81-156 Modified Newtonian theory, NASA-TND-176 Camber: AIAA-2003-3668 (Option 2 only)
CN-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124
Cm-potential	Option 1: Nose-cylinder: MBB charts, TN-WE-2-9769 Boattail: NSWC charts, NSWC-TR-81-156 Flare: Army charts AMCP 706-280, or Option 2: Slender Body Theory Camber: AIAA-2005-4971 (Option 2 only)	Option 1 and Option 2: Second Order Shock Expansion, NSWC-TR-81-156 Van Dyke Hybrid theory, NSWC-TR-81-156 Modified Newtonian theory, NASA-TND-176 Camber: AIAA-2005-4971 (Option 2 only)
Cm-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 AEDC-TR-75-124
CLL (elliptic body)	AIAA 2007-3936	AIAA 2007-3936
CA-skin friction	Turb: Van Driest II, MDAC West Handbook Lam: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom section 4.1.5.1	Turb: Van Driest II, MDAC West Handbook Lam: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom section 4.1.5.1
CA-pressure/ wave	$M < M_{crit}$ : USAF Datcom section 4.2.3.1 $M > M_{crit}$ : Cubic Fairing between to $M = 1.2$	Second Order Shock Expansion, NSWC-TR-81-156 Van Dyke Hybrid theory, NSWC-TR-81-156 Modified Newtonian theory, NASA-TND-176
CA-base	Cylinder: NSWC charts, NSWC-TR-92/509 Boattail: NASA method, NASA-TR-R-100 Flare: NSWC charts, NSWC-TR-81-358	Cylinder: NSWC charts, NSWC-TR-92/509 Boattail: NASA method, NASA-TR-R-100 Flare: NSWC charts, NSWC-TR-81-156
CA-protuberance	$M < 0.6$ : Hoerner Fluid Dynamic Drag $M > 0.6$ : cubic fairing, AIAA-94-0027	$M < 5.0$ : Modified Newtonian theory, AIAA-94-0027 $M > 5.0$ : Modified Newtonian theory
CM, CLN- protuberance	AIAA-2009-0907	AIAA-2009-0907
CA- $\alpha$	Allen and Perkins Crossflow, NASA TR-1048 AIAA 2005-4833	Second Order Shock Expansion, NSWC-TR-81-156 Assumed zero for Van Dyke Hybrid and Modified Newtonian theory AIAA 2005-4833
Dynamic derivatives	LMSC code, LMSC-D646354 and D646354A Slender Body Theory, AIAA 97-2280	LMSC code, LMSC-D646354 and D646354A Slender Body Theory, AIAA 97-2280
Magnus derivatives	SPIN 73 code, FRL-TR-4588	SPIN 73 code, FRL-TR-4588
Plume effects	not calculated	Chapman Korst model, AIAA 90-0618

Table 30. Body Alone Subroutines

Parameter	Subsonic/Transonic ( $M < 1.2$ )	Supersonic ( $M > 1.2$ )
CN-potential	Option 1: Nose-cylinder: BDCNAN, BDCNP Boattail: BDCNAB, BDCNP Flare: BDCNAF, BDCNP Option 2: SUBPTS, BDCNP Camber: CAMBOD	Option 1 and Option 2: Second Order Shock Expansion: SOSE, BDCNP Van Dyke Hybrid theory: VANDYK, BDCNP Modified Newtonian theory: HYPERS, BDCNP Camber: CAMBOD
CN-viscous	CDCS, GETETA, BDCNV	CDCS, GETETA, BDCNV
Cm-potential	Option 1: Nose-cylinder: BDXXCPN, BDCMP Boattail: BDXXCPB, BDCMP Flare: BDXXCPF, BDCMP Option 2: SUBPTS, BDCMP Camber: CAMBOD	Option 1 and Option 2: Second Order Shock Expansion: SOSE, BDCMP Van Dyke Hybrid theory: VANDYK, BDCMP Modified Newtonian theory: HYPERS, BDCMP Camber: CAMBOD
Cm-viscous	CDCS, GETETA, BDCMV	CDCS, GETETA, BDCMV
CLL (elliptic body)	ELROLL	ELROLL
CA-skin friction	SKINF, CAFRIC, BODYCA	SKINF, CAFRIC, SUPBOD
CA-pressure/ wave	$M < M_{crit}$ : BDCAPR, BODYCA $M > M_{crit}$ : CDPRES, BODYCA	Second Order Shock Expansion: SOSE, SUPBOD Van Dyke Hybrid theory: VANDYK, SUPBOD Modified Newtonian theory: HYPERS, SUPBOD
CA-base	BDCAB, BODYCA	BDCAB, SUPBOD
CA-protuberance	CAPROT, BODYCA	CAPROT, BODYCA
CM-protuberance	PROTCM	PROTCM
CLN-protuberance	PROTCM	PROTCM
CA-angle of attack	BDCALP, BODYCA	SOSE, SUPBOD
Dynamic derivatives	BDAMP, DAMP2	BDAMP, DAMP2
Magnus derivatives	SPIN83, DAMP2	SPIN83, DAMP2
Plume effects		BOTCNM, BOTCA, BASPRS

Table 31. Fin Alone Aerodynamic Methodology

Parameter	Subsonic ( $M < 0.8$ )	Transonic ( $0.8 < M < 1.4$ )	Supersonic ( $M > 1.4$ )
Airfoil Section	ADDFL-TR-71-87	$M < M_{crit}$ : AFFDL-TR-71-87 $M > M_{crit}$ : not calculated	not calculated
CN-potential	USAF Datcom 4.1.3.2 AIAA-2007-3937	RAS Data Sheets AIAA-2007-3937	$\Lambda > 0$ : USAF Datcom 4.1.3.2 $\Lambda < 0$ : AFWAL-TR-84-3084 AIAA 84-0575 AIAA-2007-3937
CN-viscous	$\Lambda > 0$ : USAF Datcom 4.1.3.3 $\Lambda < 0$ : AFWAL-TR-84-3084	$\Lambda > 0$ : USAF Datcom 4.1.3.3 $\Lambda < 0$ : AFWAL-TR-84-3084	$\Lambda > 0$ : USAF Datcom 4.1.3.3 $\Lambda < 0$ : AFWAL-TR-84-3084
XCP at $\alpha=0$	$\Lambda > 0$ : USAF Datcom 4.1.4.2 $\Lambda < 0$ : AFWAL-TR-84-3084	$\Lambda > 0$ : USAF Datcom 4.1.4.2 $\Lambda < 0$ : AFWAL-TR-84-3084	$\Lambda > 0$ : USAF Datcom 4.1.4.2 $\Lambda < 0$ : AFWAL-TR-84-3084 AIAA-2007-3937
XCP vs $\alpha$	AIAA-2007-3937	AIAA-2007-3937	AIAA-2007-3937
YCP	$M > 0.4$ : AIAA-91-0708 $M < 0.4$ : $M=0.4$ value used Flaps: AIAA-2007-3937	Tri-service data base: AIAA-91-0708 Flaps: AIAA-2007-3937	Tri-service data base: AIAA-91-0708 Flaps: AIAA-2007-3937
Flap ( $\alpha/\delta$ )	USAF Datcom section 6.1.4.1	cubic fairing	NACA-TR-1041
CA-skin friction	MDAC West Handbook Hoerner Fluid Dynamic Drag USAF Datcom section 4.1.5.1	MDAC West Handbook Hoerner Fluid Dynamic Drag USAF Datcom section 4.1.5.1	MDAC West Handbook Hoerner Fluid Dynamic Drag USAF Datcom section 4.1.5.1
CA-pressure	Hoerner Fluid Dynamic Drag	Hoerner Fluid Dynamic Drag	not applicable
CA-wave	not applicable	$M < 1.05$ : 0 $1.05 < M < 1.4$ : linear fairing	Potential Flow Theory, NWL-TR-3018
CA-bluntness	USAF Datcom section 4.1.5.1	USAF Datcom section 4.1.5.1	Potential Flow Theory, NWL-TR-3018
CA-base	Empirical, NWL-TR-2796	Empirical, NWL-TR-2796	Empirical, NWL-TR-2796
CA-induced	USAF Datcom section 4.1.5.2	USAF Datcom section 4.1.5.2	0

Table 32. Fin Alone Subroutines

Parameter	Subsonic ( $M < 0.8$ )	Transonic ( $0.8 < M < 1.4$ )	Supersonic ( $M > 1.4$ )
Airfoil Section	THEORY, CLMAX	THEORY, CLMAX	
CN-potential	FCNASB, FCNA	FCNATR, FCNA	FCNASP, FCNA
CN-viscous	FCNAAS, FCNAA	FCNAAT, FCNAA	FCNAAH, FCNAA
XCP at $\alpha=0$	$\Lambda > 0$ : FALCP, FINXAC $\Lambda < 0$ : FWDXAC, FINXAC	$\Lambda > 0$ : FALCP, FINXAC $\Lambda < 0$ : FWDXAC, FINXAC	$\Lambda > 0$ : FALCP, FINXAC $\Lambda < 0$ : FWDXAC, FINXAC
XCP vs $\alpha$	ALPXCP	ALPXCP	ALPXCP
YCP	YCP, YCP0, PANLCN	YCP, YCP0, PANLCN	YCP, YCP0, PANLCN
Flap ( $\alpha/\delta$ )	FLAPS	FLAPS	FLAPS
CA-skin friction	SKINF, CAFRIC, FINXCA	SKINF, CAFRIC, FINXCA	SKINF, CAFRIC, FINXCA
CA-pressure	FINCAP, FINXCA	FCAWT, FINXCA	
CA-wave		FINXCA	FCAWPF, FINXCA
CA-bluntness	FCALE, FINXCA	FCALE, FINXCA	FCAWPF, FINXCA
CA-base	FINCAB, FINXCA	FINCAB, FINXCA	FINCAB, FINXCA
CA-induced	FCALP, FINXCA	FCALP, FINXCA	

Table 33. Inlet Aerodynamic Methodology

Parameter	Subsonic ( $M < 1.0$ )	Supersonic ( $M > 1.0$ )
CN-potential	Engineering method, AIAA 90-3091	Engineering method, AIAA 90-3091
CN-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124
Cm-potential	Engineering method, AIAA 90-3091	Engineering method, AIAA 90-3091
Cm-viscous	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124	Jorgensen viscous crossflow, NASA-TR-R-474 and AEDC-TR-75-124
YCP		
CA-skin friction	Turb.: Van Driest II, MDAC West Handbook Lam.: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom 4.1.5.1	Turb.: Van Driest II, MDAC West Handbook Lam.: Blasius, Hoerner Fluid Dynamic Drag Roughness: USAF Datcom 4.1.5.1
CA-pressure/ wave	$M < M_{crit}$ : USAF Datcom 4.2.3.1 $M > M_{crit}$ : Transonic area rule, AIAA-90-0280	Supersonic Area rule, AIAA-90-0280
CA-additive	not applicable	Engineering method, AIAA 91-0712
CA- $\alpha$	0	0

Table 34. Inlet Subroutines

Parameter	Subsonic ( $M < 1.0$ )	Supersonic ( $M > 1.0$ )
CN-viscous	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO
Cm-potential	ILTCDC, ILTCFD, ILTVIS, ILTARO	ILTCDC, ILTCFD, ILTVIS, ILTARO
Cm-viscous	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO	CNPTWO, POTAR1, ILTARO or CNPTWO, POTAR2, ILTARO or CNPAXI, POTAR3, ILTARO
YCP	ILTCDC, ILTCFD, ILTVIS, ILTARO	ILTCDC, ILTCFD, ILTVIS, ILTARO
CA-skin friction	SKINF, CAFRIC, ILTARO	SKINF, CAFRIC, ILTARO
CA-pressure/ wave	$M < M_{crit}$ : BDCAPR, ILTARO $M > M_{crit}$ : CDPRES, ILTARO	CDPRES, ILTARO
CA-additive	IAD2D, or IADAXI, ILTARO	IAD2D or IADAXI, ILTARO
CA- $\alpha$	not applicable	not applicable

Table 35. Body-Fin Synthesis Aerodynamic Methodology

Parameter	Subsonic (M<1.0)	Supersonic (M>1.0)
Body-Fin Upwash, $K_W$	Empirical correlation, AIAA 96-3395 Folding fin: AIAA 94-0027	Empirical correlation, AIAA 96-3395 Folding fin: AIAA 94-0027
Fin-Body Carryover, $K_B$	Slender body theory, NACA-TR-1307	Slender body theory, NACA-TR-1307 AIAA Journal, May 1981 AIAA-2007-3937
Body-Fin Upwash Center of Pressure, $x_{cp_{WB}}$	$\Lambda > 0$ : USAF Datcom section 4.1.4.2 $\Lambda < 0$ : AFWAL-TR-84-3084	$\Lambda > 0$ : USAF Datcom section 4.1.4.2 $\Lambda < 0$ : AFWAL-TR-84-3084
Body-Fin-Body Carryover Center of Pressure, $x_{cp_{BW}}$	Lifting line theory, NACA-TR-1307 AIAA 94-0027	Slender body theory, NACA-TR-1307 AIAA Journal, August 1982
Fin Deflection, $\Lambda_{IJ}$	Slender body theory, AGARD-R-711	Slender body theory, AGARD-R-711
Equivalent angle of attack	AIAA J. S&R, July-Aug 1983	AIAA J. S&R, July-Aug 1983
Body Vortex Strength	Empirical, NWC-TP-5761	Empirical, NWC-TP-5761
Body Vortex Track	Empirical, NWC-TP-5761	Empirical, NWC-TP-5761
Fin Vortex Strength	Line vortex theory, NACA-TR-1307	Line vortex theory, NACA-TR-1307
Fin Vortex Track	along velocity vector	along velocity vector
Dynamic derivatives	Equiv. angle of attack, AIAA 97-2280	Equiv. angle of attack, AIAA 97-2280

Table 36. Body-Fin Synthesis Subroutines

Parameter	Subsonic (M<1.0)	Supersonic (M>1.0)
Body-Fin Upwash, $K_W$	KWBNEW, SYNTHS Folding fin: PANLCN	KWBNEW, SYNTHS Folding fin: PANLCN
Fin-Body Carryover, $K_B$	CARRYO, SYNTHS	CARRYO, SYNTHS
Body-Fin Upwash Center of Pressure, $x_{cp_{WB}}$	$\Lambda > 0$ : FALCP, CARRYO, SYNTHS $\Lambda < 0$ : FWDXAC, CARRYO, SYNTHS	$\Lambda > 0$ : FALCP, CARRYO, SYNTHS $\Lambda < 0$ : FWDXAC, CARRYO, SYNTHS
Body-Fin-Body Carryover Center of Pressure, $x_{cp_{BW}}$	CARRYO, SYNTHS	CARRYO, SYNTHS
Fin Deflection, $\Lambda_{IJ}$	FINFIN, PANLCN	FINFIN, PANLCN
Equivalent angle of attack	ALPEQ, PANLCN	ALPEQ, PANLCN
Body Vortex Strength	CLVR, ALPEQ, PANLCN	CLVR, ALPEQ, PANLCN
Body Vortex Track	CLVR, ALPEQ, PANLCN	CLVR, ALPEQ, PANLCN
Fin Vortex Strength	VRINTS, SYNTHS	VRINTS, SYNTHS
Fin Vortex Track	SFWRW, SVTRAK, SYNTHS	SFWRW, SVTRAK, SYNTHS
Dynamic derivatives	ALPEQ2, FDAMP, DAMP2	ALPEQ2, FDAMP, DAMP2